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(NASA-CR-159551) JT8D REVISED HIGH-PRESSURE
TURBINE COOLING AND OTHER OUTER AIR SEAL
PROGRAM (Pratt and Whitney Aircraft Group)
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JT8D REVISED HIGH-PRESSURE TURBINE COOLING AND OUTER
AIR SEAL PROGRAM

by

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UNITED TECHNOLOGIES CORPORATION
PRATT & WHITNEY AIRCRAFT GROUP
COMMERCIAL PRODUCTS DIVISION

Prepared for

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

NASA-Lewis Research Center
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FOREWORD

The development and demonstration effort described in this report was conducted by the Commercial Products Division of Pratt & Whitney Aircraft Group, United Technologies Corporation, under NASA Contract NAS3-20630. Mr J. A. Ziemianski and Tom Strom of the NASA-Lewis Research Center were the Project Manager and Project Engineer, respectively, for the contract.

This report was prepared by Mr. William O. Gaffin, Pratt & Whitney Aircraft Program Manager with the assistance of Messrs. Albin S. Janus and Craig S. Vickery. The technical data presented herein were compiled with the cooperation of a wide segment of Engineering personnel. This report has been assigned the Commercial Products Division, Pratt & Whitney Aircraft Group Internal Report Number PWA-5515-77.

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1.0 SUMMARY

The objective of the JT8D Revised High-Pressure Turbine Cooling and Outer Air Seal Program (hereinafter referred to as the revised HPT OAS) was to demonstrate the TSFC improvement of this concept. Preliminary analysis predicted a 0.2 percent improvement at sea level takeoff, and 0.5 percent at average cruise conditions. An exhaust gas temperature (EGT) reduction of 4°C at take-off and 3°C at climb was also predicted.

To accomplish these improvements, the first-stage high-pressure turbine outer air seal and blade cooling scheme were redesigned. The abradable honeycomb seal rubbing area was increased to accommodate an additional knife edge and the existing spoiler as second and third seals. A different material was used in the seal support ring to obtain a better thermal expansion match between the turbine disk and seal. The turbine blade cooling configuration was also changed because the additional seal improvements blocked the cooling air discharge route originally used in the Bill-of-Materials blades.

Comparison tests at sea level and simulated altitude conditions were run using the Bill-of-Materials seal and blades, and revised outer air seal and blades installed in an experimental JT8D-17R engine. After the tests, the hardware was removed and inspected, and the data was reduced and analyzed. TSFC improvements of 0.61 percent at sea level takeoff and 0.60 percent at average cruise condition, and exhaust gas temperature reductions of 6°C (11°F) at takeoff and 5°C (9°F) at climb conditions were obtained with the revised turbine hardware. The revised hardware showed no unusual wear or degradation following the engine testing.

The demonstrated performance improvements are better than the estimates used in the Engine Component Improvement Program Feasibility Analysis evaluation. Thus the airline acceptability of the concept is enhanced and the estimated cumulative fuel saving (as defined by the Feasibility Analysis) is increased from 340×10^6 liters to 767×10^6 liters.

2.0 INTRODUCTION

National energy demand has outpaced domestic supply, creating an increased U.S. dependence on foreign oil. This increased dependence was dramatized by the OPEC oil embargo in the winter of 1973-74. In addition, the embargo triggered a rapid rise in the cost of fuel which, along with the potential of further increases, brought about a changing economic circumstance with regard to the use of energy. These events, of course, were felt in the air transport industry as well as other forms of transportation. As a result of these experiences, the government, with the support of the aviation industry, has initiated programs aimed at both the supply (sources) and demand (consumption) aspects of the problem. The supply problem is being investigated by looking at increasing fuel availability from such sources as coal and oil shale. An approach to the demand aspect of the problem is to evolve new technology for commercial aircraft propulsion systems which will permit development of a more energy efficient turbofan or the use of a different propulsive cycle such as a turbo-prop. Although studies have indicated large reductions in fuel usage are possible (e.g., 15 to 40 percent), the fuel savings impact of developing and introducing into service a new turbofan or turboprop engine would not be significant for at least ten to fifteen years. In the short term, the only practical propulsion approach is to improve the fuel efficiency of current engines. Examination of this approach has indicated that a five percent fuel reduction goal, starting in the 1980-82 time period, is feasible. In as much as commercial aircraft in the free world are using fuel at a rate in excess of 80 billion liters of fuel per year, even five percent represents significant fuel savings.

Since a major portion of the present commercial aircraft fleet is powered by the JT8D and JT9D engines, NASA is sponsoring a program whose objective is to reduce the fuel consumption of these engines. This program, called the Engine Component Improvement (ECI) program, has two main parts, performance improvement and engine diagnostics. The latter part, which is not reported herein, is aimed at identifying the sources and causes of engine deterioration. The performance improvement (PI) part is intended to identify and evaluate the concepts which are technically and economically viable for the 1980-82 time period, and then develop and demonstrate these concepts through ground and flight tests. The PI program was conducted by Pratt & Whitney Aircraft under NASA contract NAS3-20630. Eight promising concepts were identified from a list of over one hundred for further work and development.

The first improvement to emerge from the Pratt & Whitney Aircraft ECI-PI program is the JT8D Revised High-Pressure Turbine Cooling and Outer Air Seal concept. This concept improves the fuel consumption of the JT8D-11, -15, -17, -17R series of engines, hereinafter referred to as the JT8D. The concept achieves this improvement by reducing air seal leakage past the tip shrouds of the high-pressure turbine (HPT). This leakage reduction increases the efficiency of the HPT and the engine as a whole, improving fuel consumption. The revised cooling scheme in the HPT blades also improves the fuel consumption by decreasing cooling air mixing losses.

To evaluate the magnitude and feasibility of this improvement, Pratt & Whitney Aircraft initiated a program to design, fabricate, and test this concept and compare it with the Bill-of-Materials design currently in production. The results of this program are discussed herein.

3.0 DESIGN AND FABRICATION

3.1 SEAL DESIGN

The current JT8D-11, 15, 17, 17R high pressure turbine blade tip leakage is controlled by a single knife edge on the blade tip, running against a nickel-base abradable honeycomb strip. A significant seal leakage reduction can be achieved by adding another knife edge to the blade tip, increasing the width of the honeycomb strip to seal the added knife edge and the existing spoiler, and altering the material of the support ring from Hastelloy "C" to Hastelloy "S" to better match the thermal expansion of the disk. Figure 3-1 compares the two seal configurations.

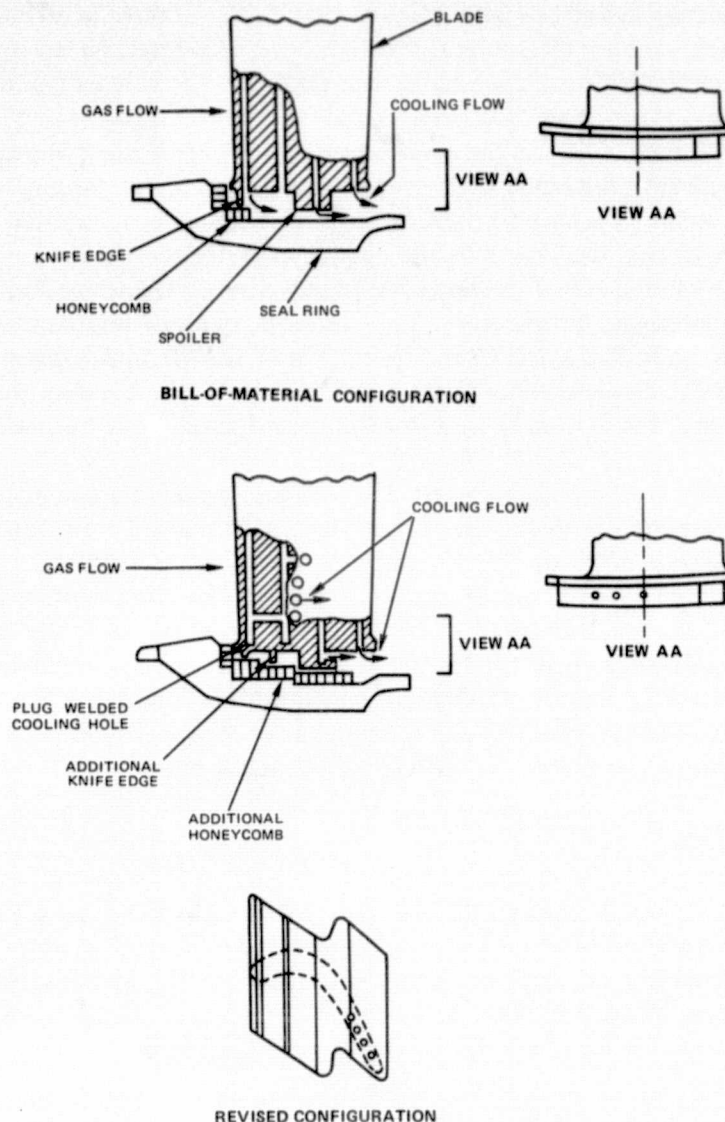


Figure 3-1 Comparison of Bill-of Materials and Revised Blade Configurations. Schematic illustrates the changes made to the revised blade. All cooling air paths are not shown, only representative paths.

Leakage reduction was maximized by stepping the knife edges on the blade tip and filling the honeycomb abradable cells 50 percent with sintered nickel chromium insulative material. The insulative material slows the response of the seal ring temperature to changes in primary gas flow temperature, thus matching the rotor disk response better. The stepped knife edges create a greater flow restriction than if they were in line with each other. Seal operating clearance was established by adjusting the cold clearance so that the sealing surfaces just touch during the critical transient condition. This condition occurs during deceleration from stabilized sea level takeoff power to idle power.

The Bill-of-Materials high-pressure turbine blade is aircooled through 11 radial holes discharging out of the tip. Incorporation of an additional shroud knife edge and utilization of the spoiler as a seal to improve outer air seal performance would restrict cooling flow in the first seven holes, resulting in higher metal temperatures and a decrease in airfoil life. In order to maintain the present cooling flow several revisions have been made to the airfoil cooling scheme. The first four leading edge holes have been vented to the convex (suction) side of the airfoil and the next three holes have been vented to the rear of the spoiler as shown in Figure 3-2. The tip discharge holes in the revised blade were plug welded in order to prevent ingestion of gaspath air into the blade with a subsequent reduction in cooling flow. Increased flow restriction due to the new cooling path through the blade is compensated by the decreased exit pressure, so that the flow rate and cooling capacity remain the same. Venting of the cooling air to the suction side does, however, reduce flow to the tip region of the airfoil and consequently increases its temperature. The increased tip metal temperature is not expected to reduce the life of the airfoil since it is still below the life limiting temperature, as shown in Figure 3-3. The increased tip metal temperature is also less than the current peak blade temperature, which is unchanged by the revised cooling configuration.

Redirection of the blade cooling discharge air also results in a performance improvement. Approximately half the cooling air for the revised blade is discharged from four holes on the suction surface (Figure 3-4). The Mach number in this region is lower than at the blade tip trailing edge, reducing the mixing loss for this portion of the blade cooling air compared to what it would be if it were discharged at the tip. There is also an advantage in that the suction side discharge direction is closer to that of the mainstream gas, which also reduces the mixing loss (Figure 3-5). For the JT8D high-pressure turbine there is a one to one correspondence between total pressure loss and high-pressure turbine efficiency. Therefore, any decrease in mixing loss can be directly related to an efficiency improvement.

3.2 FABRICATION

The hardware for the JT8D improved HPT blade tip seal was fabricated by reworking a set of semifinished Bill-of-Material turbine blade castings and machining a new outer air seal from a non-Bill-of-Material Hastelloy "S" forged ring. One hundred blades were obtained from production. These blades had finished roots and airfoils, and partly finished tips. The blades were sent to a vendor for reworking to the new configuration.

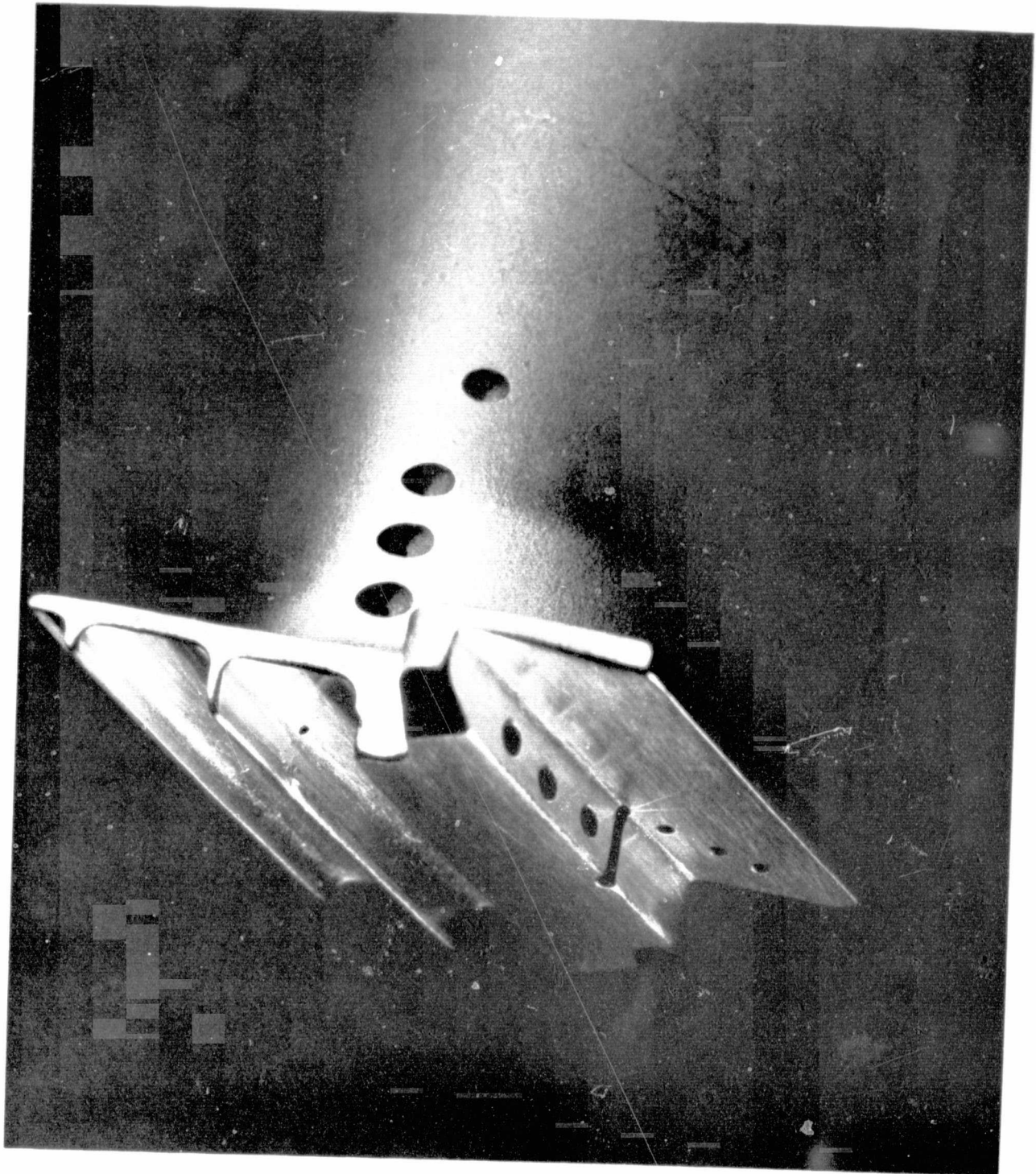


Figure 3-2 Revised Blade With New Cooling-Air Discharge Locations.

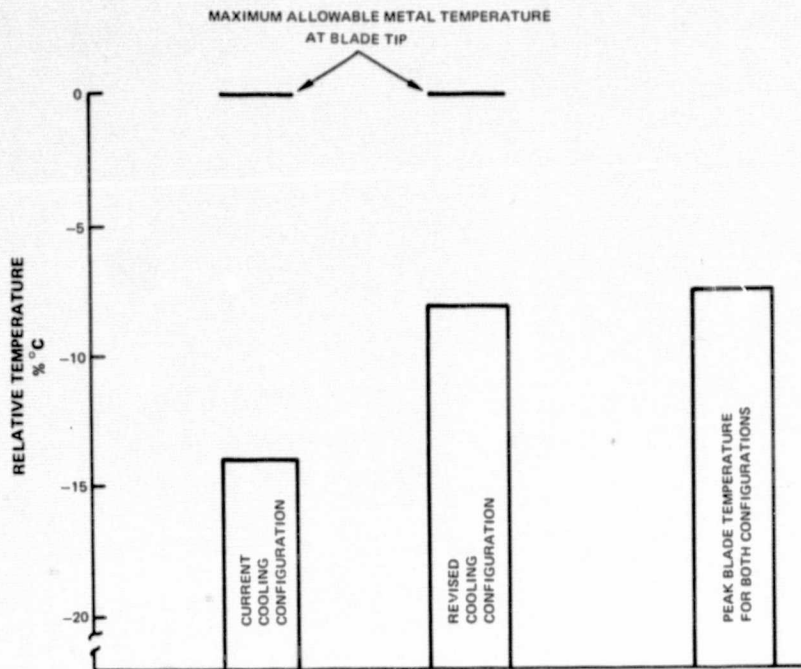


Figure 3-3 Blade Life Limiting Temperature for the Bill-of-Materials and Revised HPT OAS at Blade Tip

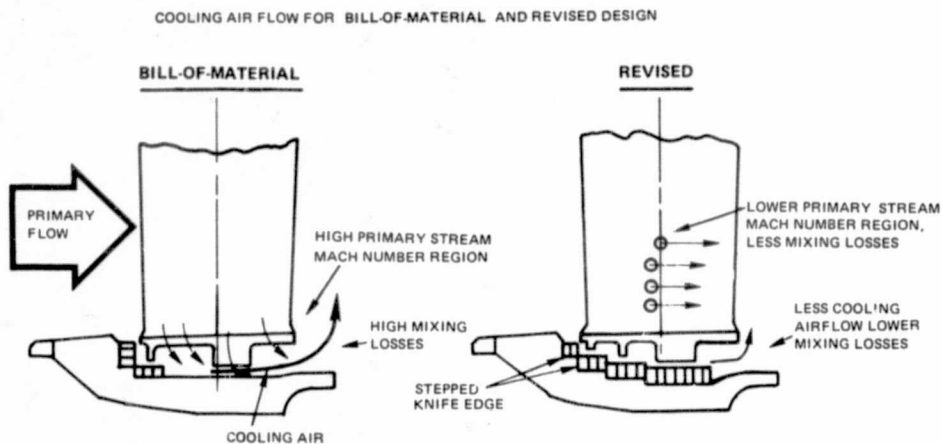


Figure 3-4 Pressure Mixing Losses for the Bill-of-Material and Revised Blade Configuration. The revised blade cooling configuration reduces mixing losses.

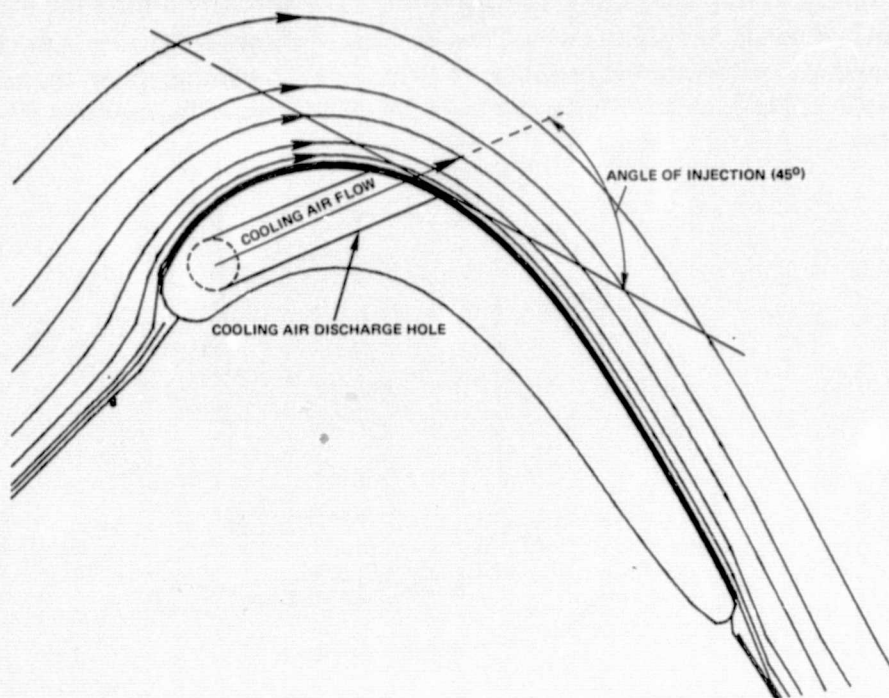


Figure 3-5 Streamline Plot for the Revised Blade Configuration at 90.9% Span. The cooling air is injected at a closer angle to the primary flow, reducing mixing losses.

Four cooling air discharge holes were electrical discharge machined into the suction side of the airfoil and three discharge holes were electrical discharge machined into the rear of the spoiler. The blade tip shroud was then machined to 0.025 cm (0.010 inches) over finished thickness and the original holes and the blade shroud notches were weld filled. After welding, the blades were finish machined to blueprint requirements.

The HPT outer air seal was made from a special forging of Hastelloy "S", because this material provides a better thermal response than Hastelloy "C", which is used in the Bill-of-Materials seal. Since the seal was a new configuration, new honeycomb rings were fabricated by a vendor while the air seal support ring was being machined in Pratt & Whitney Aircraft's Experimental Machine Shop. The ring and honeycomb seal lands were then sent to an outside vendor to be brazed and to have the honeycomb cells partially filled with sintered nickel chromium insulation.

The finished air seal had a gap in the honeycomb steps as a result of a dimension change between preliminary design and final design, as shown in Figure 3-6. However, this gap was filled with insulating material, and calculations indicated that the knife edges would not run over the gap and affect sealing. Examination of the hardware after testing confirmed this.

Some minor cracking was encountered in the blade shrouds as a result of weld plugging operations. The blades were accepted despite these cracks, since previous experience with similar shroud cracks indicated that they would not propagate during the planned test program and would not affect the test results. This conforms to Quality Assurance procedures applying to experimental programs (Appendix A). A photograph of the cracks is shown in Figure 3-7. The problem will be eliminated in production with improved fabrication techniques.

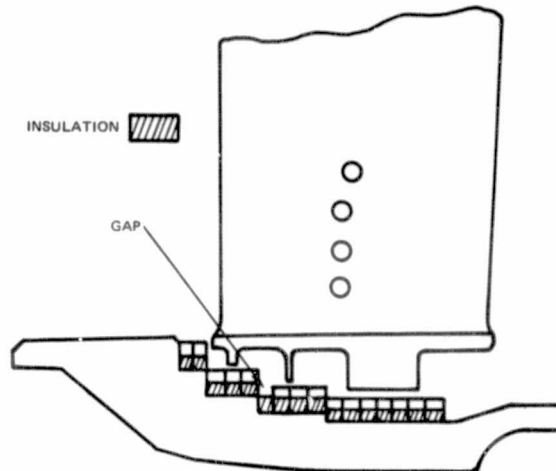


Figure 3-6 Location of Gap in the Honeycomb Seal.

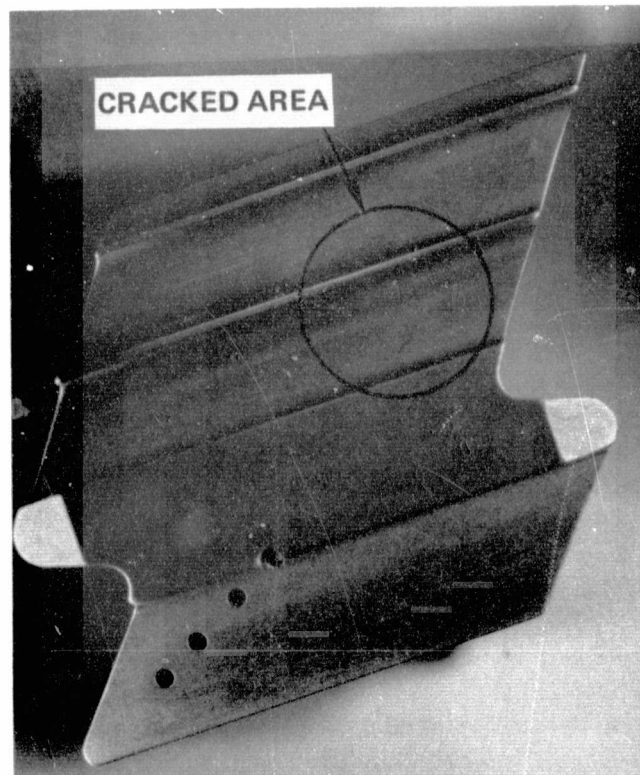


Figure 3-7 Photograph of Cracks Caused by Plug Welding of Cooling Holes

When the blades were assembled in the disks, it was discovered that the circumferential tip shroud clearances on the revised blades were more open than on the Bill-of-Materials blades. Test data existed that could be used to correct for the performance effects of the difference, so it was decided to proceed to avoid a long delay in the program while new blades were procured. Measurement of the shrouds of the revised blades showed they had been machined as much as 0.011 cm (0.0045 inches) undersize in the area shown on Figure 3-8. This discrepancy apparently resulted from run-out of the finish grinding operation that was performed after the shroud notches were plug welded.

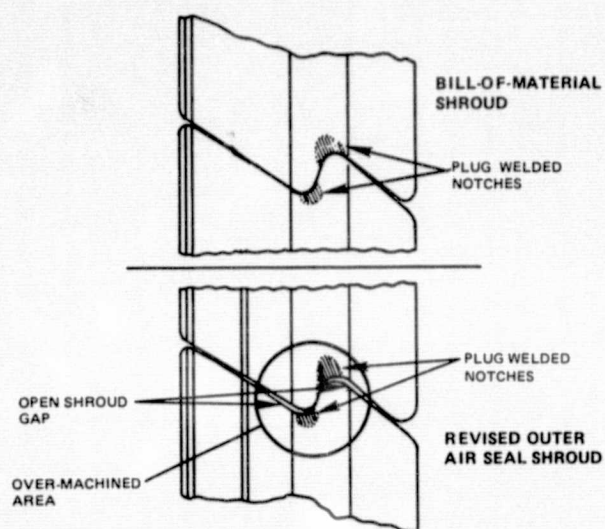


Figure 3-8 Undersize Dimension On Blade Shrouds

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4.0 TEST FACILITY AND INSTRUMENTATION

4.1 TEST VEHICLE

The test vehicle used in both the sea level test and the simulated altitude test was engine X-372, built to approximate the performance of a Bill-of-Materials JT8D-17R production engine. This engine had demonstrated good performance characteristics during the year before the test. It is instrumented to a much greater degree than production or other development engines. The engine was mounted in test stand X-16 for the sea level test and test stand X-209 for the altitude test.

4.2 X-16 SEA LEVEL TEST STAND

X-16 stand is a gas turbine engine test facility designed to develop both afterburning and non-afterburning turbojet and turbofan engines. Total airflow in the stand is limited by the inlet sound treatment to 817 kg/sec (1800 lb/sec). Engine testing can be conducted at static sea level inlet and exhaust conditions.

The stand is constructed of reinforced concrete in the form of an elongated "L". A horizontal inlet and a resonant chamber exhaust silencer with a vertical discharge stack are located at the extreme ends. Acoustical panels with 42 percent open area are installed in the inlet. The test engine is attached to a suspended overhead thrust measurement platform. Engine total airflow is measured by a bellmouth inlet. The stand inlet is isolated from the after part of the stand by a partial bulkhead. Exhaust gases from the engine are ejected into and mixed with atmospheric air that has been drawn around the partial bulkhead.

The controls and instrumentation necessary to operate the engine and monitor its performance are located in the test stand control room. This room is located between this stand and X-15 stand. An observation window is provided in the control room for visual inspection of the test cell interior during engine operation. Test stand support equipment and services are located beneath the control room.

4.3 X-209 ALTITUDE TEST STAND

X-209 is a full-scale engine stand in the Willgoos Laboratory designed to test engines at simulated altitude conditions. The stand consists of an enclosed test cell 27 m long x 7.6 m high (90 ft x 25 ft) above the grating and an adjacent air conditioned control room 7.8 m x 8.2 m x 3 m high (26 ft x 27 ft x 10 ft). The test cell contains an altitude chamber 3.7 m (12 ft) in diameter and 10 m (34 ft) long within which the test engine is mounted on a moveable thrust measurement platform. The engine inlet is sealed to a 3.7 m (12 ft) diameter inlet chamber, while the engine exhaust is ejected through the opposite end of the chamber. Air required to operate the test engine can be drawn directly from the atmosphere or it can be supplied under pressure from the laboratory air compressor units. In addition, inlet air can be cooled to as low as -48°C (-55°F) by passing the air through the laboratory air refrigeration system, or heated to as high as 330°C (625°F) by passing the air through two air heater units located adjacent to the test cell. Engine inlet airflow is mea-

sured with choked nozzles. Up to nine nozzles are available in the X-209 stand. Engine exhaust gas can be discharged directly to the atmosphere or to the laboratory exhauster and within the test chamber, thereby simulating operation of the engine at various altitudes. Through controlled use of the laboratory compressors, air heaters and refrigeration systems, in coordination with the exhauster units, altitude and flight speed conditions can be simulated over a wide range.

4.4 INSTRUMENTATION

Both X-16 and X-209 test stands are incorporated in the P&WA Steady State Data System (SSDS) which is controlled by an on-line computer system. The test stands are within separate subsystems on the SSDS, and data acquisitions are placed in a queue when more than one SSDS stand is running simultaneously.

The SSDS will accept three basic types of inputs: electro-motive forces (EMF's), pneumatic pressures, and frequencies. These inputs, on both the X-16 and the X-209 test stands, total 949 channels with approximately 822 of the channels available to record engine test stand instrumentation. There are 357 inputs available for EMF measurements, 576 for pneumatic pressure and 40 for frequency signals. These basic inputs are used to measure:

- Low pressure rotor speeds
- High pressure rotor speeds
- Thrust
- Airflow
- Fuel flow
- Temperatures
- Pressures
- Dew point

A detailed list is contained in Table 4-1. Engine instrument locations are shown in Figure 4-1.

Special instrumentation was installed in the high-pressure turbine of the test vehicle for assessing the effectiveness of the new seal design. This instrumentation consisted of twelve thermocouples and eighteen static pressure taps. Eight thermocouples were buried below the inner and outer surfaces of the support ring. This instrumentation was located in two circumferential locations and in selected axial locations (see Figure 4-2). Static pressure taps were located in the air cavities of the high pressure turbine case and in the air cavity formed by the second vane and the low pressure turbine case (see Figure 4-3). This special instrumentation was used to measure pressure in the area of the revised seal and measure temperatures around the seal for clearance calculations.

TABLE 4-1
CALIBRATION INSTRUMENTATION

ARP* Station	Eng. Location	Parameter Symbol	Description	ARP* Station	Eng. Location	Parameter Symbol	Description
0.0	Bellmouth Screen	PT0	4 Kiel Probes~Total Pressure	5.0	Behind LPT	PT7	1 Manifolded Reading from 6 Probes with 6 Samples per Probe ~Total Pressure
		TT1	12 Calibrated Thermocouples (C T/C's)~Total Temperature			PS7	1 OD Reading at Each PT7 Probe Location~Static Pressure
1.0	Foreward of IGV	PT2, PS2	6 Pitot Static Probes~Total and Static Pressure			TT7	8 Rakes with 1 Average per Rake ~Total Temperature
13.0	Behind FEGV	PT2.5	4 Pole Rakes with 5 Readings per Rake~Total Pressure		External Edge of Tail Pipe	PS JET	4 Static Pressure Taps
		TT2.5	16 Individual Temp. Probes C T/C's~Total Temperature		Front of Test Stand Bulkhead	PS FBH	4 Static Pressure Taps
2.2	Behind LPC	PT3	4 Rakes with 5 Readings per Rake ~Total Pressure		Rear of Test Stand Bulkhead	PS RBH	4 Static Pressure Taps
		PS3	2 Bleed Cavity Readings~Static Pressure		Engine External Skin Pressure	PS SKIN	3 Locations Along Engine with 2 Readings per Location~ Static Pressure
		TT3	4 Rakes with 5 Readings per Rake ~Total Temperature	16.0	Rear of Fan Duct	PT7F	1 Manifolded Reading From 6 Probes with 6 Samples per Probe ~Total Pressure
3.0	Behind HPC	PT4	3 Rakes with 4 Readings per Rake ~Total Pressure		Fuel Flow	Wf	2 In-Line Turbine Meters
		PS4	2 Bleed Cavity Readings~Static Pressure			TT COX 1, TT COX 2	2 Fuel Temperature Probes
		PB	1 Fuel Control Pressure Sense ~Static Pressure		Thrust	FN	2 Strain Gage Load Cells
		TT4	2 Rakes with 4 Readings per Rake~Total Temperature		Low Rotor Speed	N1	1 ILS Tachometer
4.0	Turbine Cooling Air Static Pressure	PCP	1 Bleed Cavity Pressure		High Rotor Speed	N2	1 ILS Tachometer
	Turbine Outer Air Seal (TOAS)	TTTOAS	6 CT/C's Imbeded in 1st TOAS				
4.1	L.E. 2nd Turbine Nozzle Vane	PT6	6 Vane L.E. Clusters of 3 Vanes Each with 5 Readings per Cluster ~Total Pressure				
		PS6	1 OD Reading at Each PT6 Cluster Location~Static Pressure				

*As defined by SAE

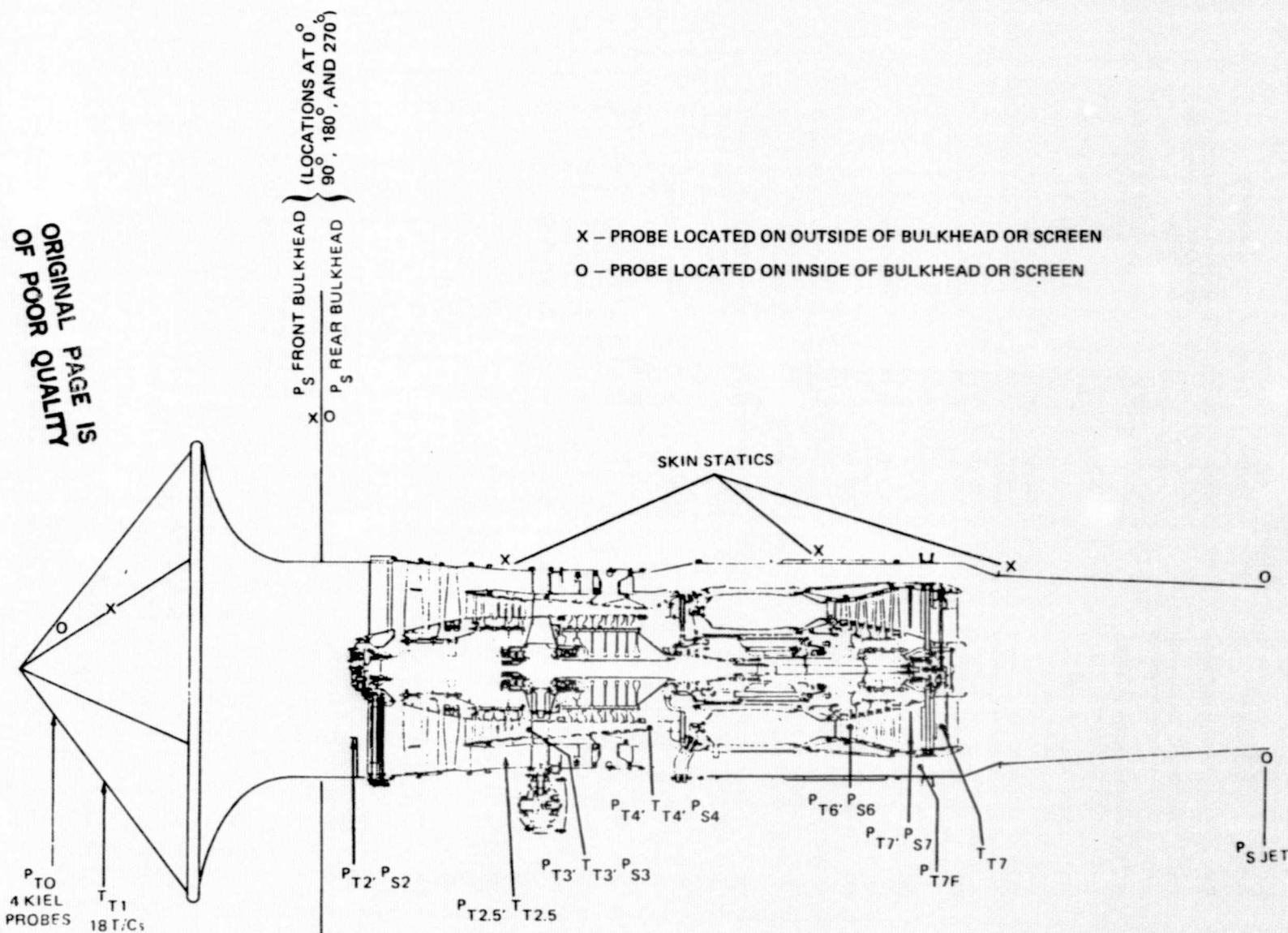


Figure 4-1

Engine Instrumentation Locations. P_{TO} , T_{T1} , and Bulkhead Pressure are Measured in the X-16 Stand. Engine internal pressures and temperatures were measured in both the X-16 and X-209 stands.

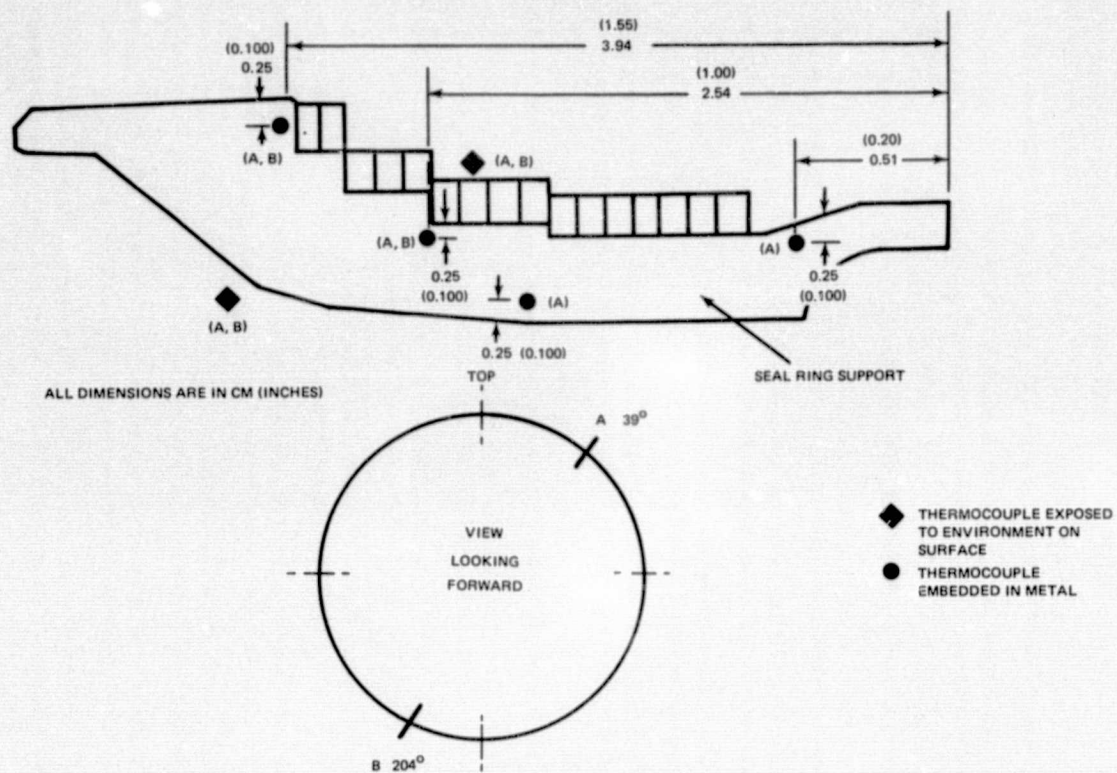


Figure 4-2 Thermocouple Locations for the OAS

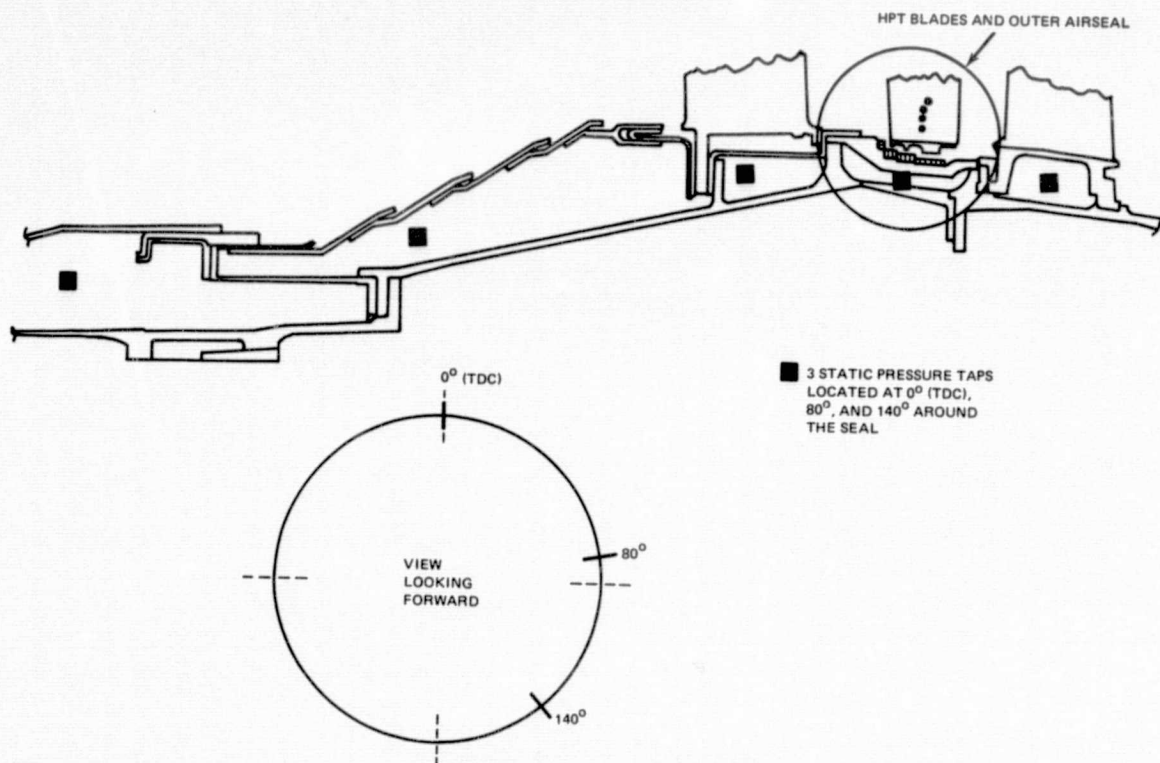


Figure 4-3 Static Pressure Tap Locations for the OAS

5.0 TEST PROCEDURE

5.1 SEA LEVEL AND ALTITUDE TEST

The performance testing was conducted on the X-16 stand sea level stand and X-209 altitude stand (as described in Sections 4.2 and 4.3) to define the difference in TSFC between the Bill-of-Materials configuration and the revised configuration. Engine gas generator parameters were used to determine the effect of the configuration change independent of the thrust/TSFC measurement and to reduce uncertainty.

The engine was mounted in the stand and all instrumentation installed and calibrated. Fuel meters were calibrated prior to installation. The thrust meter was calibrated after engine and instrumentation installation and hook up to ensure that the thrust bed would be free to move. The engine was started and a run made to check out its operation, i.e., oil and fuel systems, stand connections, instrumentation, vibration and data recording and reduction systems. This procedure was used on both X-16 and X-209 stands and is typical of most installations.

Two ten point calibrations were run for both the Bill-of-Material and revised configurations, starting at high power. The altitude test sequence was modified to maximize facility use by running the 50 percent maximum cruise data point first. The raw data was corrected for measurement inconsistencies, i.e., thermocouple wire calibrations, etc. and converted to engineering units. The data was provided to the "quick look" program, (see Figure 5-1) for real time processing and to tape for later processing. The "quick look" program calculates engine performance in abbreviated form within five minutes of data acquisition, and is used for checking data quality during testing and preliminary performance assessments. The follow on computer program provides editing and profiling of the data through interactive terminals, more detailed computer processing, print output, data plotting and curve fitting.

The test procedures and requirements were as follows:

A thrust meter adjustment calibration was done prior to testing and after any remount or module change. "As is" thrust meter checks were taken before and immediately after each performance calibration while the stand was still warm to ensure that no significant changes had occurred in the thrust measurement system.

Fuel samples were taken before and immediately after each sea level and altitude performance calibration. The samples were sent to the Materials Engineering and Research Laboratory for specific gravity, lower heating value, and fuel viscosity analyses. A fuel sample was taken during a calibration if a long shutdown was required.

A check was made of instrumentation and data recording systems at idle and at 44,000 newtons (10,000 pounds) observed thrust at sea level and an equivalent altitude rating prior to the acquisition of performance data. Each performance calibration power setting was stabilized before the acquisition of data. Discrepant items were corrected. A calibration was not conducted when critical instrumentation was inoperative.

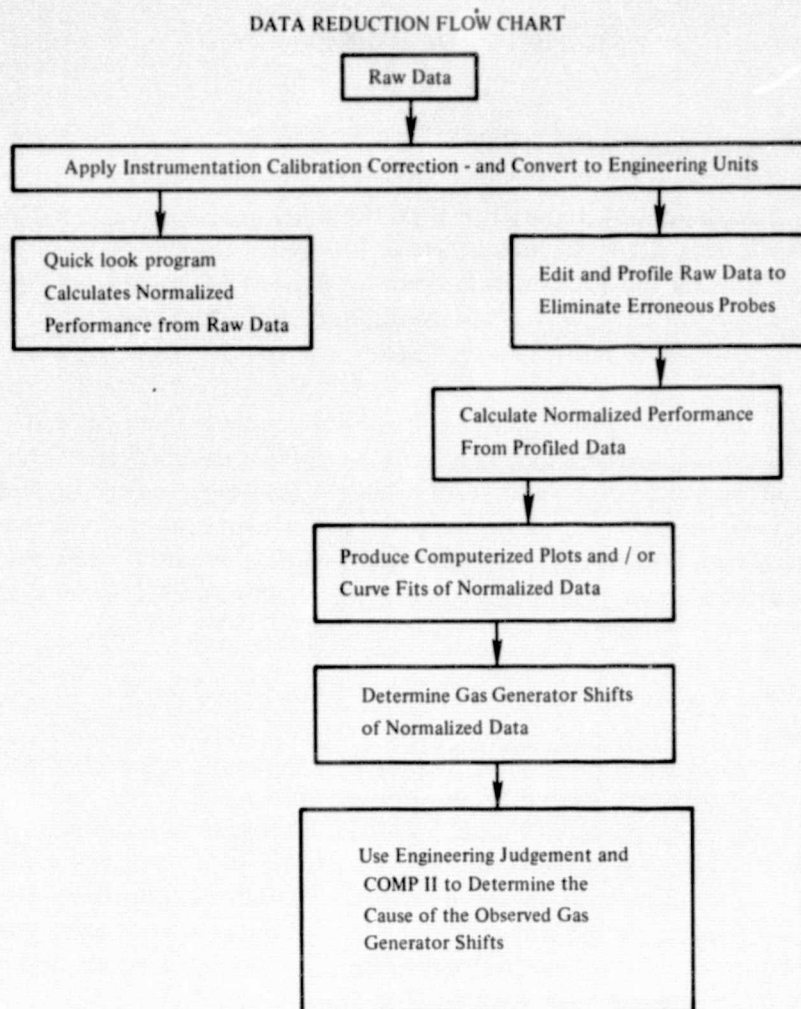


Figure 5-1 Data Reduction Flowchart

The two 1.91 cm (3/4-inch) turbine type fuel flow meters were calibrated prior to their use. Post-test "as is" calibrations were also done. The same two meters were used throughout the test sequence for both the X-16 and X-209 stands.

Prior to the sea level calibration with each seal configuration, a hot snap deceleration was conducted to wear-in the seal, consistent with production acceptance test procedures (Figure 5-2). The steady state sea level calibration consisted of two sets of ten power setting points run in decreasing power sequence. These ten points are defined in Table 5-1.

Before running the simulated altitude test calibration a calibration was run in X-209 stand as close to sea level as possible in order to check out the stand operation and engine instrumentation. The steady state altitude calibration consisted of two sets of ten power setting points at 9100 meters (30,000 feet), Mach number 0.8 simulated conditions, run in decreasing power sequence. An intermediate power setting point is run first to check stand systems and their effect on data. The ten points are defined in Table 5-2.

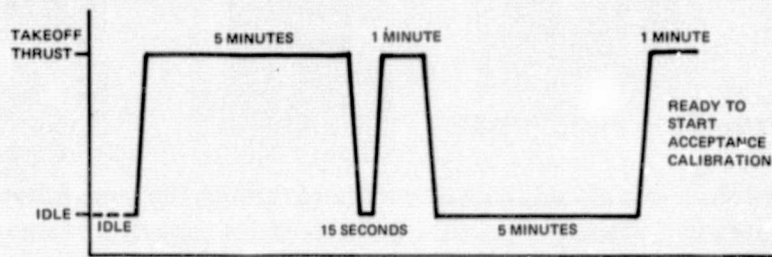


Figure 5-2 Production Acceptance Snap Accelerations and Decelerations Were Run to Wear in Seals Throughout the Engine.

TABLE 5-1

STEADY-STATE SEA LEVEL CALIBRATION

Point No.	Corrected Net Thrust newtons (pounds)
1	76,000 (17,000)
2	71,000 (16,000)
3	67,000 (15,000)
4	58,000 (13,000)
5	49,000 (11,000)
6	44,000 (10,000)
7	40,000 (9,000)
8	27,000 (6,000)
9	13,000 (3,000)
10	Idle

TABLE 5-2

STEADY-STATE ALTITUDE CALIBRATION

Point No.	Corrected Low-Pressure Rotor Speed (rpm)
1	6800 (50% max cruise)
2	8500 (105% max cruise)
3	8300 (100% max cruise)
4	8000 (90% max cruise)
5	7700 (80% max cruise)
6	7400 (70% max cruise)
7	7100 (60% max cruise)
8	6500 (40% max cruise)
9	6200 (30% max cruise)
10	4900 (idle)

5.2 CALCULATION PROCEDURE

The sea level and simulated altitude data are processed through the same follow-on computer program. The data which is peculiar to one type of stand (i.e., sea level vs simulated altitude) is handled in separate program subroutines. These subroutines include the sea level stand thrust corrections, and the simulated altitude stand calculations for inlet momentum, engine buoyancy, altitude and Mach number as run, and adjustment to the desired altitude and Mach number. The measured thrust, fuel flow, rotor speeds, pressures, and temperatures are processed by this computer program to provide gross and net corrected thrust, corrected fuel flow, pressure ratios, component efficiencies, and airflow, including total, engine and duct flows.

Thrust specific fuel consumption was determined from fuel mass flow rates and net thrust measurements. A computer program processes the raw data obtained from the test, as described previously, to produce corrected gas generator parameters and adjusted altitude, thrust, and fuel flow measurements. A flow chart of the analysis process is given in Figure 5-1.

Thrust was measured with a strain gage load cell mounted to the engine and the testbed. The measured thrust was adjusted to the actual centerline thrust, and inlet air property variations were taken into account. Inlet air momentum and ram effect were taken into account for the altitude test. Engine and test frame buoyancy was found to be negligible for the altitude thrust calculation.

Fuel flow was determined using two fuel flow turbine meters. Volumetric flow rate can be calculated from the rotational speed of the meter. Fuel density measurements were made by the Material Lab and were used to calculate the fuel mass flow rate from the volumetric flow rate.

When the calculated TSFC for the two configurations is compared at constant thrust, the TSFC difference between them is due to the component change. This component change also causes shifts in other gas generator parameters. These gas generator differences are compared to expected differences from influence coefficients. Since the component change is expected to increase high turbine efficiency, the gas generator parameter changes at constant engine pressure ratio (EPR) would be expected to increase overall compressor pressure ratio and high and low rotor speeds, and high pressure compressor (HPC) discharge temperature with very small increases in total airflow and LPC discharge temperature. A decrease in EGT and LPC pressure ratio would be expected. These parameter shifts are used to confirm the measured TSFC difference using a computer program. This P&WA computer program, the Multi-variant Comprehensive Gas Generator Statistical Analysis program (COMP II), statistically combines the measured gas generator shifts, the instrumentation uncertainty and the influence coefficients and defines an expected TSFC difference and an uncertainty. Influence coefficients for several likely changes are input and the program will operate on all those input and indicate which of the changes are the most likely.

6.0 RESULTS

6.1 PERFORMANCE RESULTS

The thrust specific fuel consumption (TSFC) improvement demonstrated for the revised HPT OAS program was determined from data derived from fuel flow rates and net thrust measurements. The TSFC improvement includes a correction factor for the excessive shroud gap clearance that was unique to the revised OAS blades. This TSFC improvement was compared to TSFC calculations derived from a statistical gas generator analysis of test data (COMP II) and an analytical approach based on turbine section temperature and pressure data. Although the three methods summarized in Figure 6-1 yield somewhat different levels of improvement, the gas generator analysis and analytical approach lend credence to the performance improvement based on measured thrust and fuel flow data.

All engine exhaust gas temperature (EGT) reductions discussed in this section were derived from the statistical gas generator analysis.

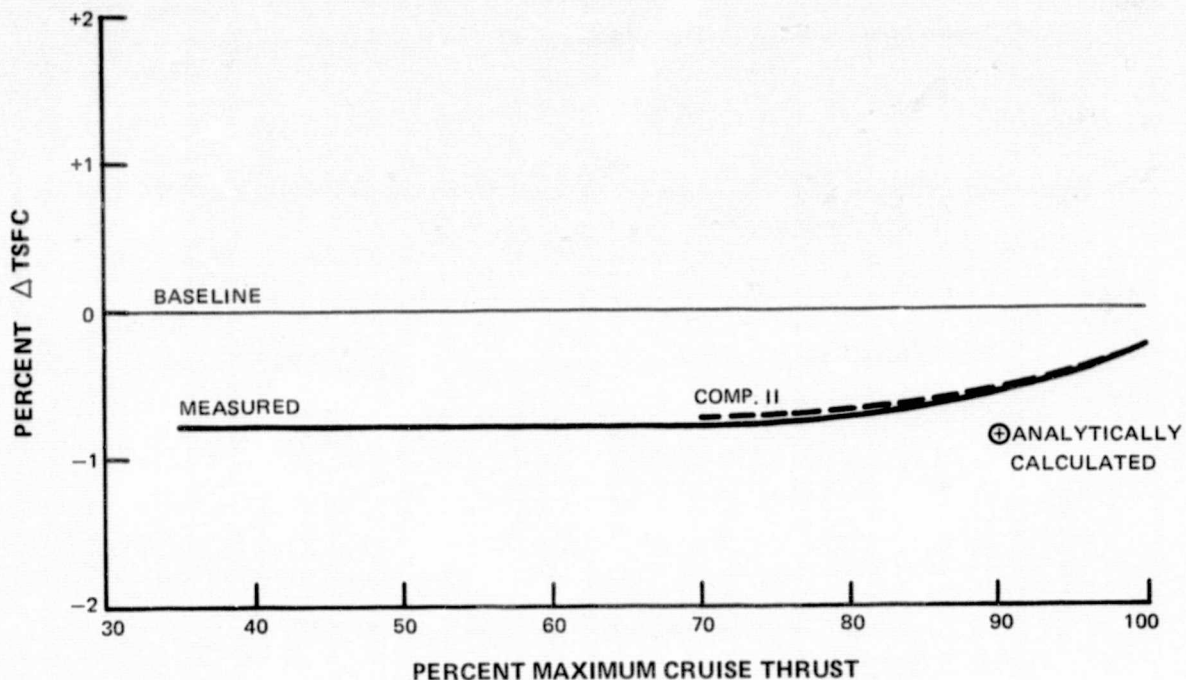


Figure 6-1 Comparison of Baseline, COMP II, Measured and Analytically Calculated TSFC Improvement Versus Percent Maximum Cruise Thrust at 9,100 m (30,000 ft) Mach Number 0.8 Simulated Condition

Measured Data

The revised OAS configuration improved TSFC 0.61 percent at sea-level takeoff and 0.60 percent at 90 percent maximum cruise. The TSFC improvement as a function of thrust is plotted in Figures 6-2 and 6-3 for sea level and cruise conditions, respectively.

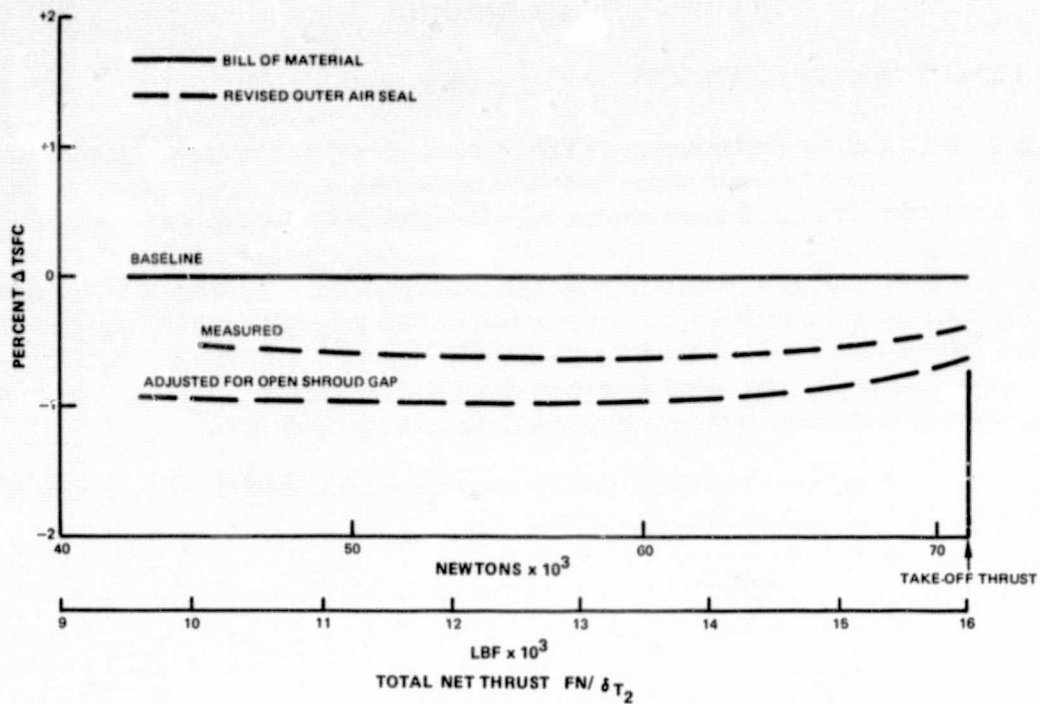


Figure 6-2 Change In TSFC From Baseline for Measured and Adjusted Results At Sea Level Conditions

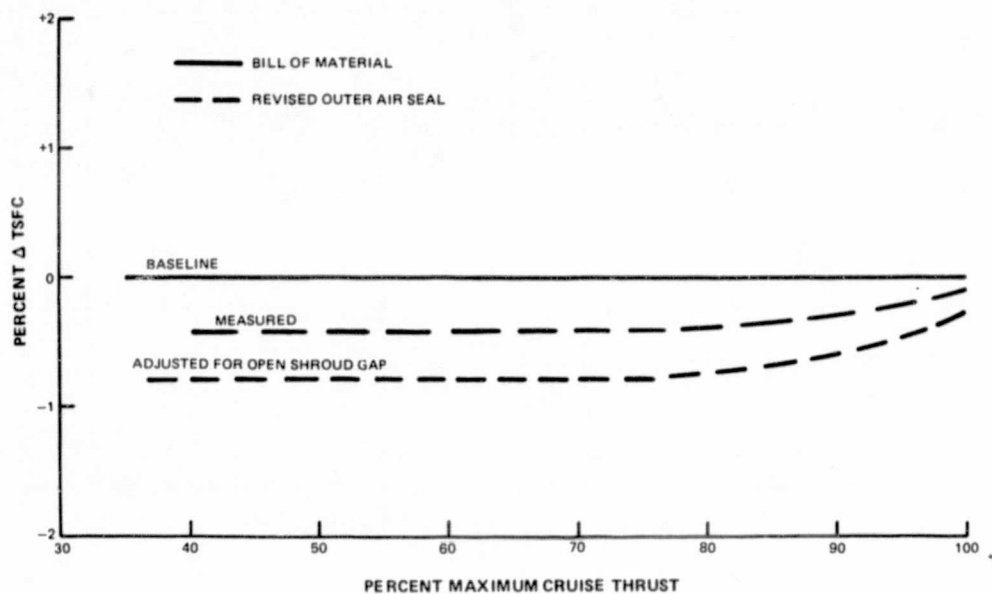
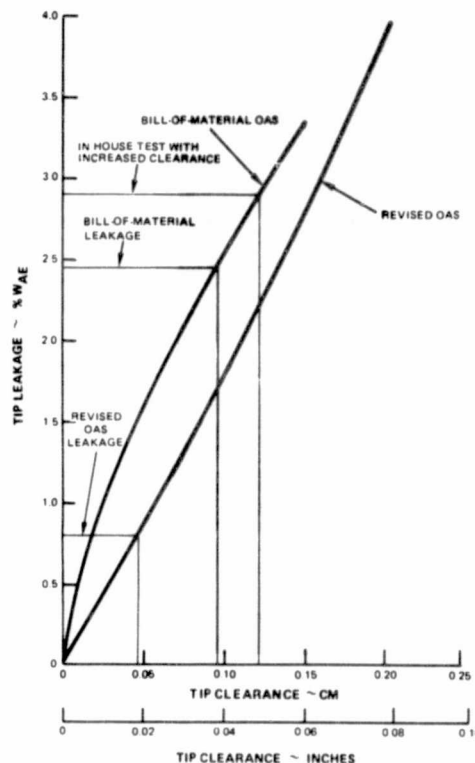


Figure 6-3 Change In TSFC From Baseline for Measured and Adjusted Results At 9,100 m (30,000 ft) Mach Number 0.8 Simulated Conditions

An inspection of the revised OAS blades, as discussed in Section 3.2, showed that the tip shroud circumferential clearances were greater than the Bill-of-Materials clearances. These increased clearances for the revised OAS would be expected to have an adverse effect on the performance of this configuration. The previously described measured results include a correction factor to account for this difference. This correction factor was determined from previous test data obtained from a turbine blade shroud test. In that test, the effect of sealed HPT blade notches on performance was investigated. A set of Bill-of-Materials blades was modified so that the notches were completely sealed. Comparison testing between the modified and Bill-of-Material configurations revealed no difference in TSFC. However, the modified configuration had a blade tip to seal clearance of 0.0254 cm (0.010 inches) greater than the Bill-of-Material configuration. The cold flow rig leakage curve for the Bill-of-Material OAS (Figure 6-4) was used to extract the tip leakage reduction that would result if this clearance was brought back to the Bill-of-Material specifications. This was determined to be 0.46 percent and translates directly into a HPT efficiency gain of 0.46 percent. Therefore a HPT efficiency loss of 0.46 percent can be equated to the turbine blade shroud notch area prior to sealing. The difference in the area of the shroud gap experienced in the revised OAS testing was 1.3 times that of the turbine shroud notch area test. Scaling of the shroud notch area to the excessive shroud gap of the revised OAS revealed that the HPT efficiency for the revised OAS configuration should increase 0.60 percent if the shroud gap is reduced to Bill-of-Material clearances. This translates into a TSFC improvement of 0.24 percent at sea level takeoff and 0.30 percent at 90 percent maximum cruise.



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Figure 6-4 Cold Flow Rig Test Leakage as a Function of Clearance for the Bill-of-Materials and Revised HPT OAS

The actual measured TSFC improvement was 0.37 percent at sea level static at takeoff throttle setting and 0.30 percent at an altitude of 9100 meters (30,000 ft.) and Mach number of 0.80, and 90 percent of maximum cruise throttle setting. The back-to-back test results of the revised OAS configuration are shown in Figures 6-5 and 6-6 where the TSFC change is shown as a function of thrust for sea level static and altitude cruise conditions respectively.

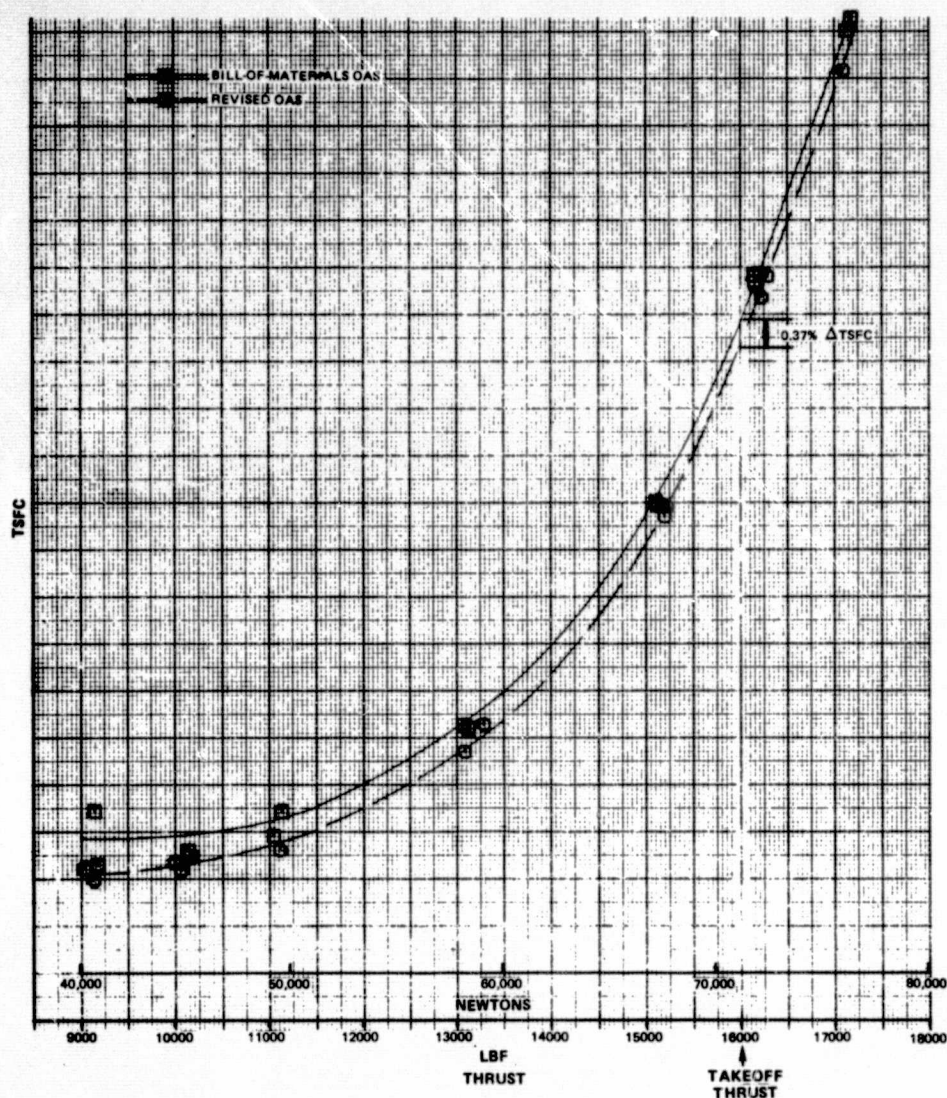


Figure 6-5 TSFC Curve Showing Data Points and Curve Fit for Sea Level Conditions

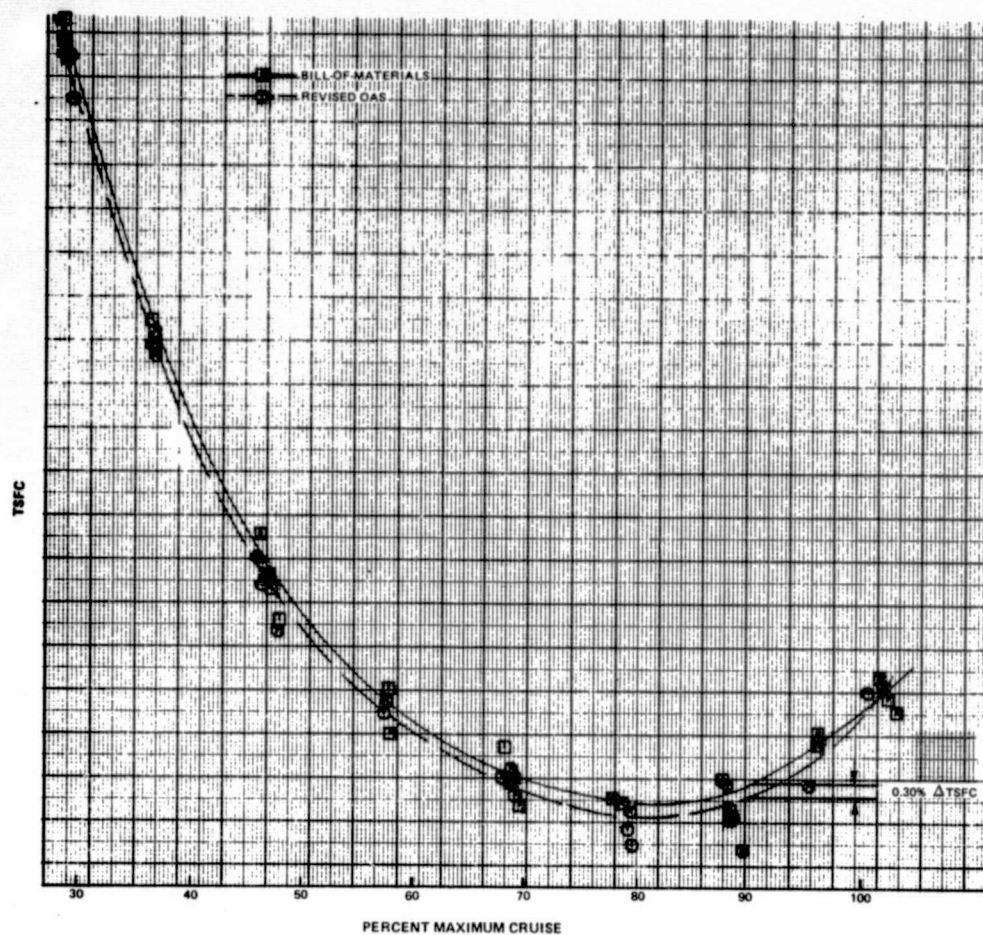


Figure 6-6 TSFC Curve Showing Data Points and Curve Fit for 9100 m (30,000 ft) Mach Number 0.8 Simulated Conditions

Gas Generator Analysis

A P&WA computer program, the Multivariant Comprehensive Gas Generator Statistical Analysis Program (COMP II), was used to calculate a TSFC improvement that could be compared with the measured results. This program combined the measured gas generator parameter shifts shown in Figures 6-7 to 6-20, the instrumentation uncertainty and the influence coefficients to define the TSFC differences and uncertainties shown on Figures 6-21 and 6-22. These results agree well with the uncorrected results from the thrust and fuel flow measurements shown on Figures 6-2 and 6-3. The results of the (COMP II) analysis, in terms of turbine efficiency, TSFC, and EGT improvements are summarized on Table 6-1 for the two representative operating conditions. This table also shows the corrections for excessive shroud gap clearances based on the analysis described above.

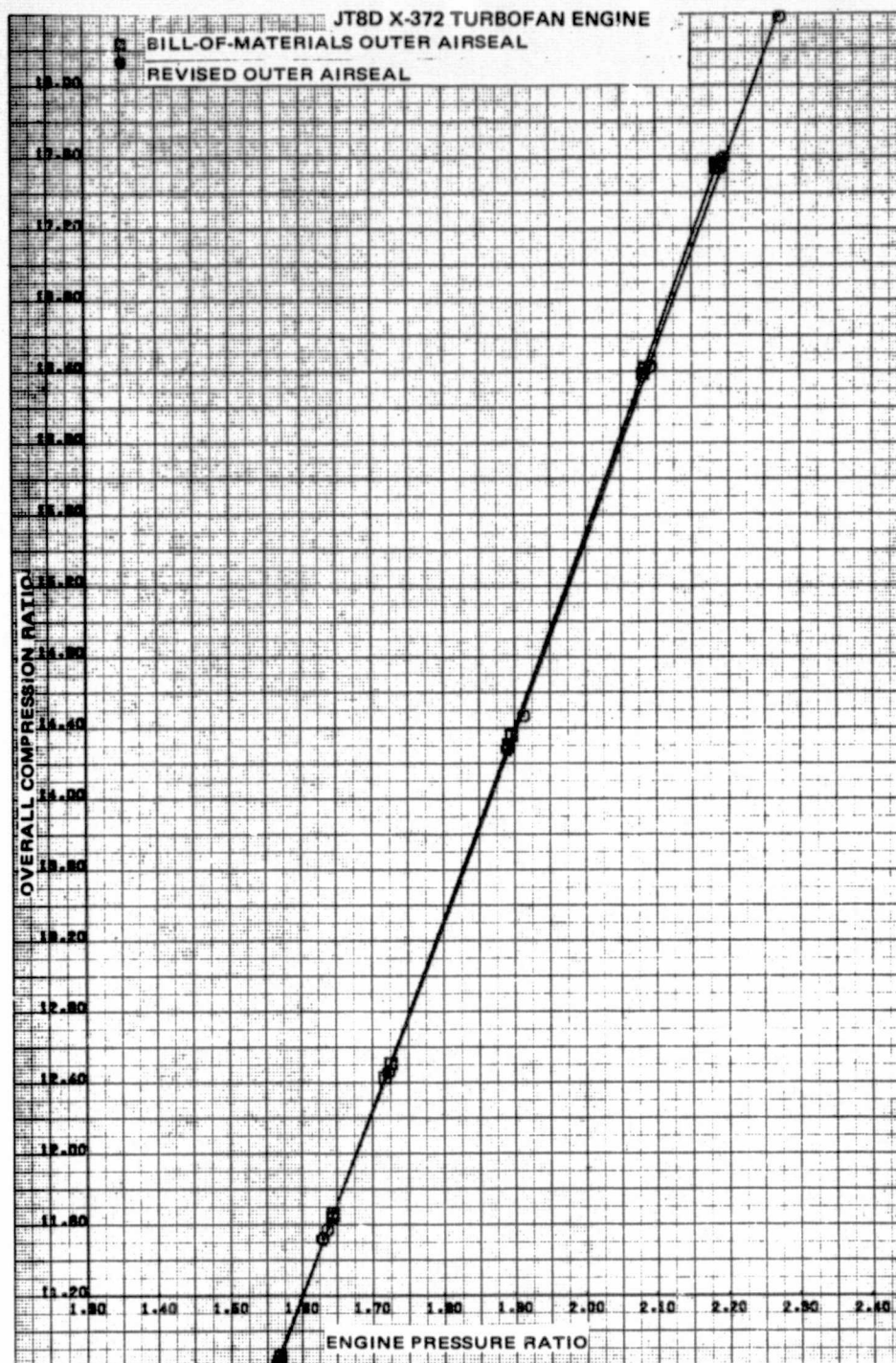


Figure 6-7 Overall Compression Ratio vs Engine Pressure Ratio for the Bill-of-Materials and Revised HPT OAS At Sea Level.

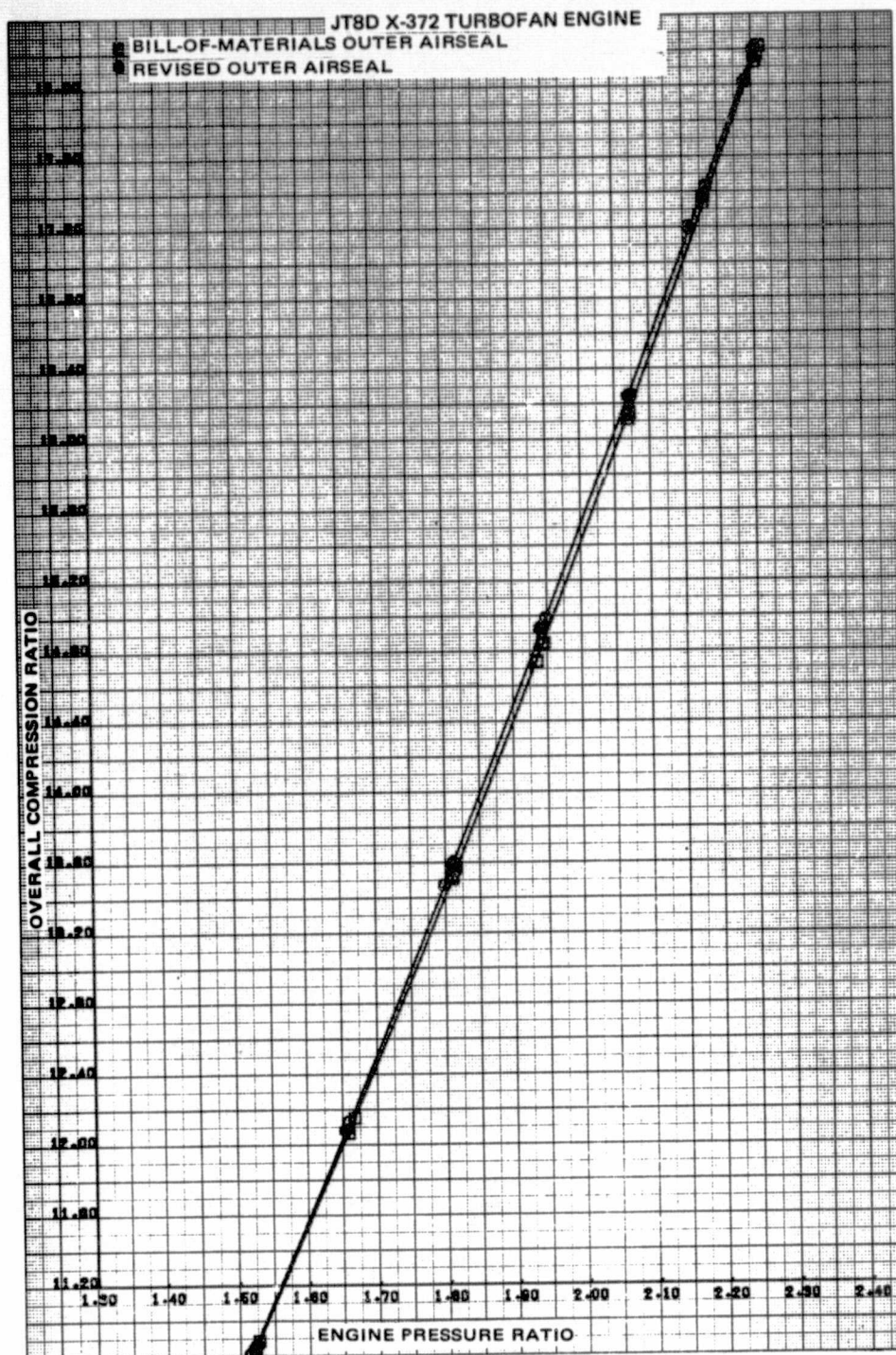


Figure 6-8 Overall Compression Ratio vs Engine Pressure Ratio for the Bill-of-Materials and Revised HPT OAS At 9,100 m (30,000 ft.) Mach Number 0.8 Simulated Conditions.

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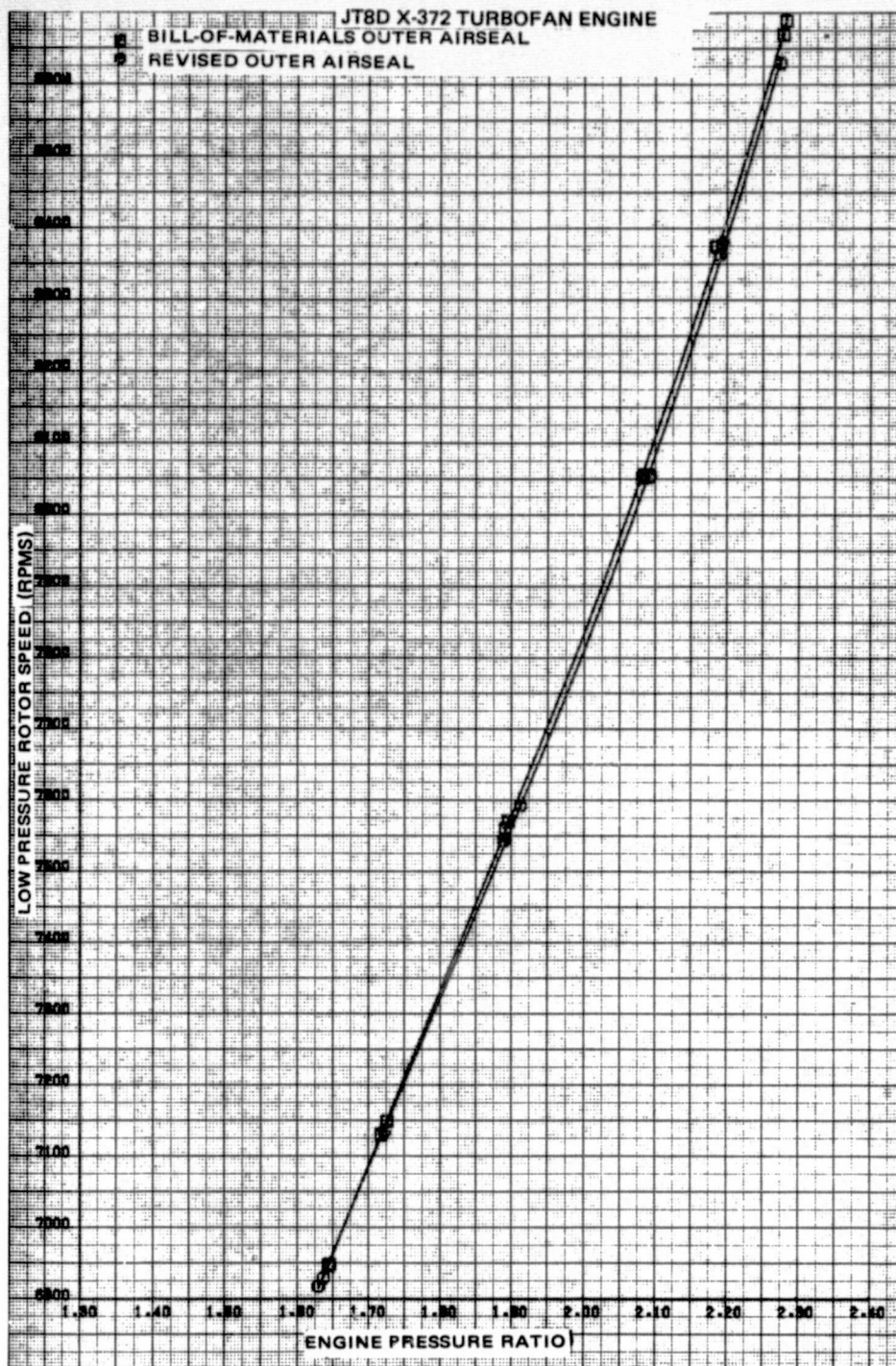


Figure 6-9 Low Pressure Rotor Speed vs Engine Pressure Ratio for the Bill-of-Materials and Revised HPT OAS At Sea Level.

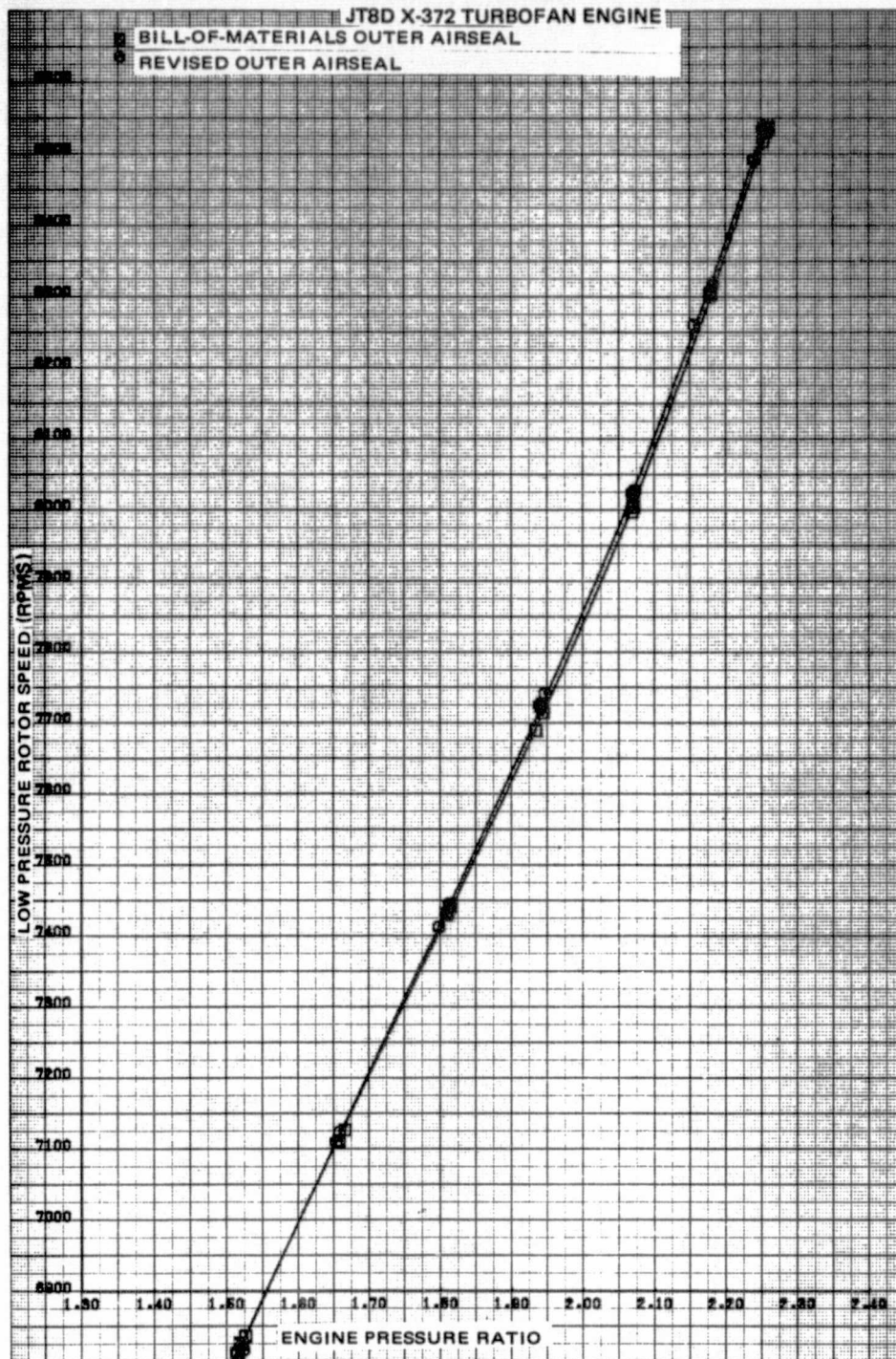


Figure 6-10 Low Pressure Rotor Speed vs Engine Pressure Ratio for the Bill-of-Materials and Revised HPT OAS At 9,100 m (30,000 ft.) Mach Number 0.8 Simulated Conditions.

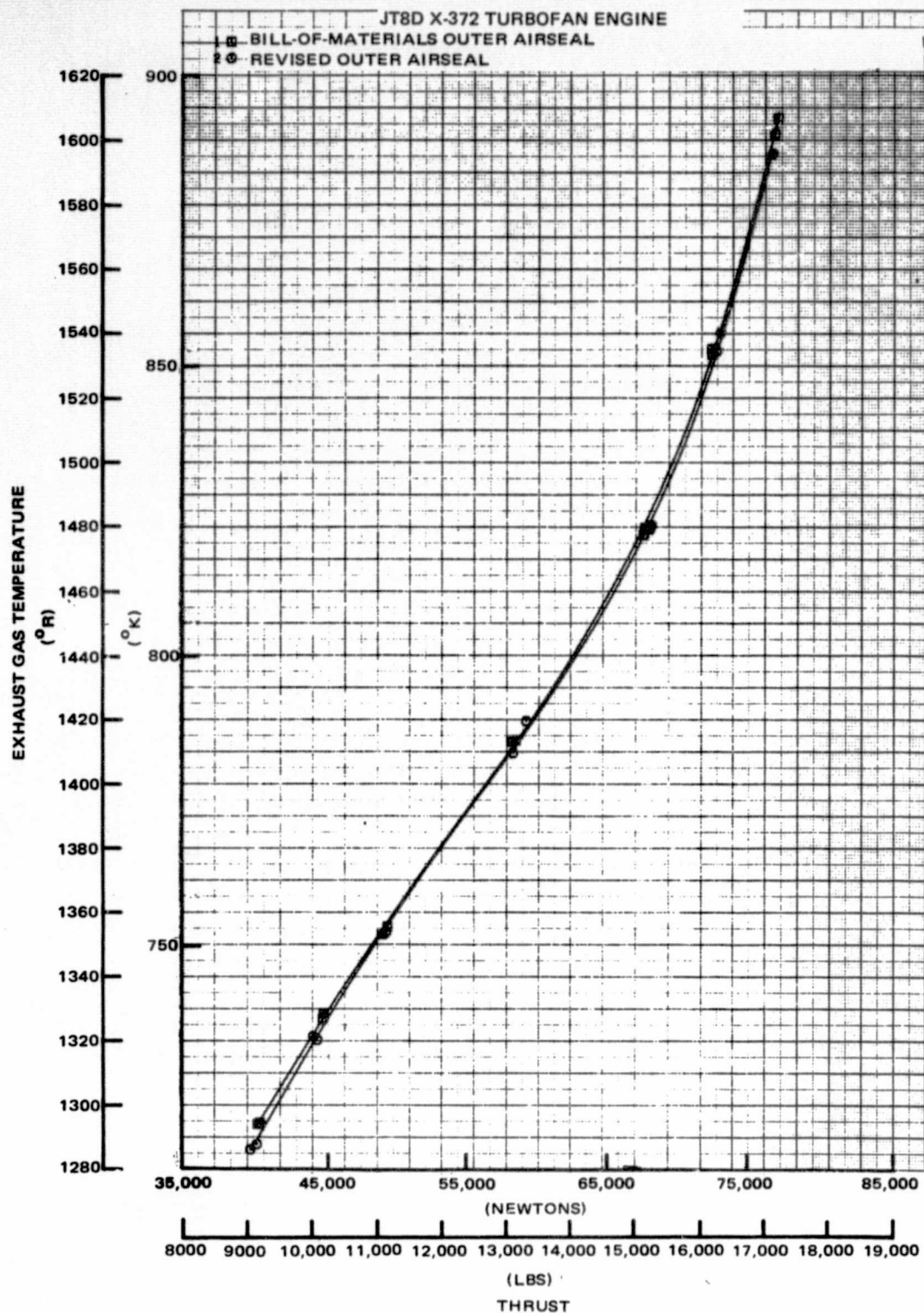


Figure 6-11 Exhaust Gas Temperature vs Thrust for the Bill-of-Materials and Revised HPT OAS At Sea Level.

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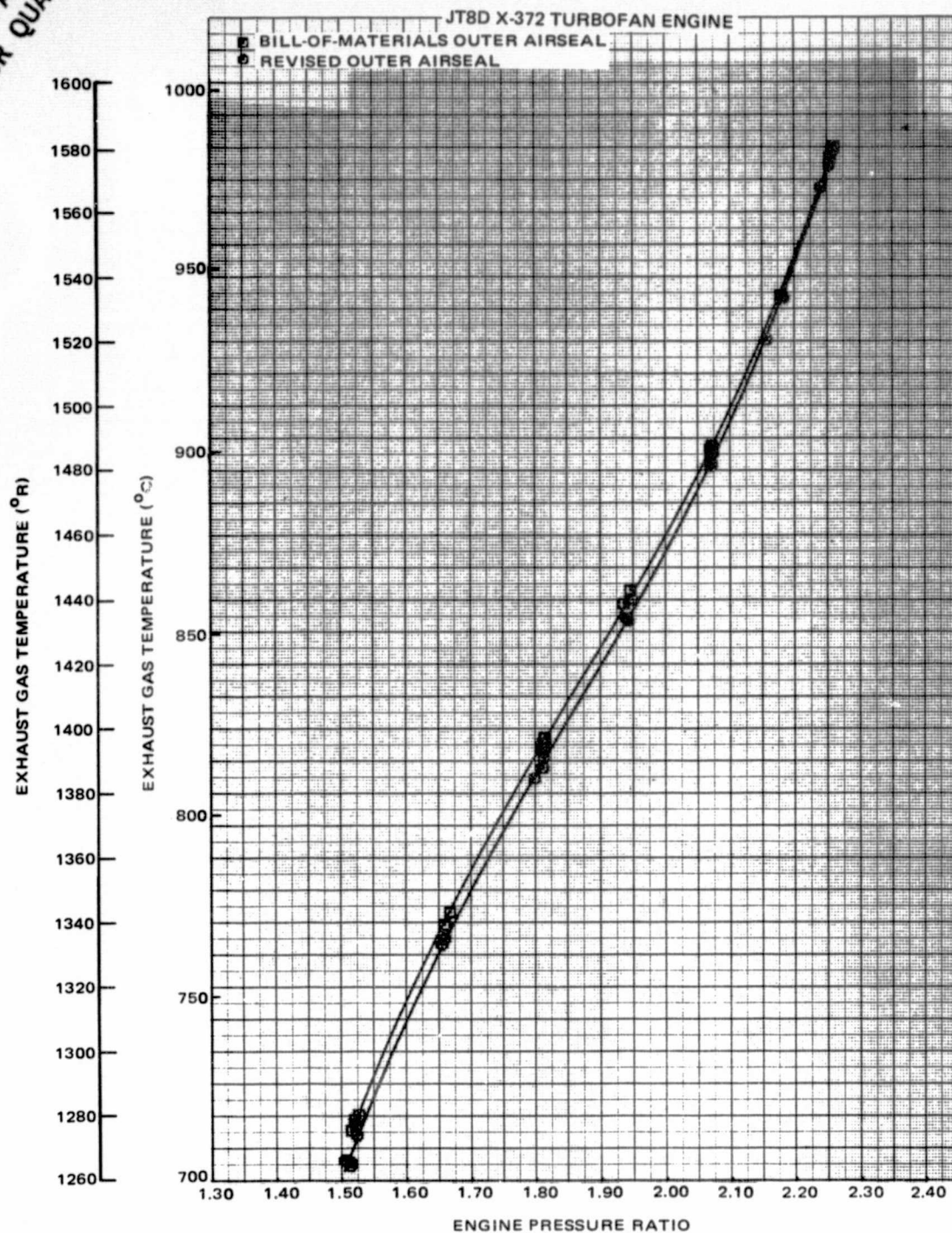


Figure 6-12 Exhaust Gas Temperature Vs. Engine Pressure Ratio for the Bill-of-Materials and Revised HPT OAS at 9,100 m (30,000 ft) Mach Number 0.8 Simulated Conditions

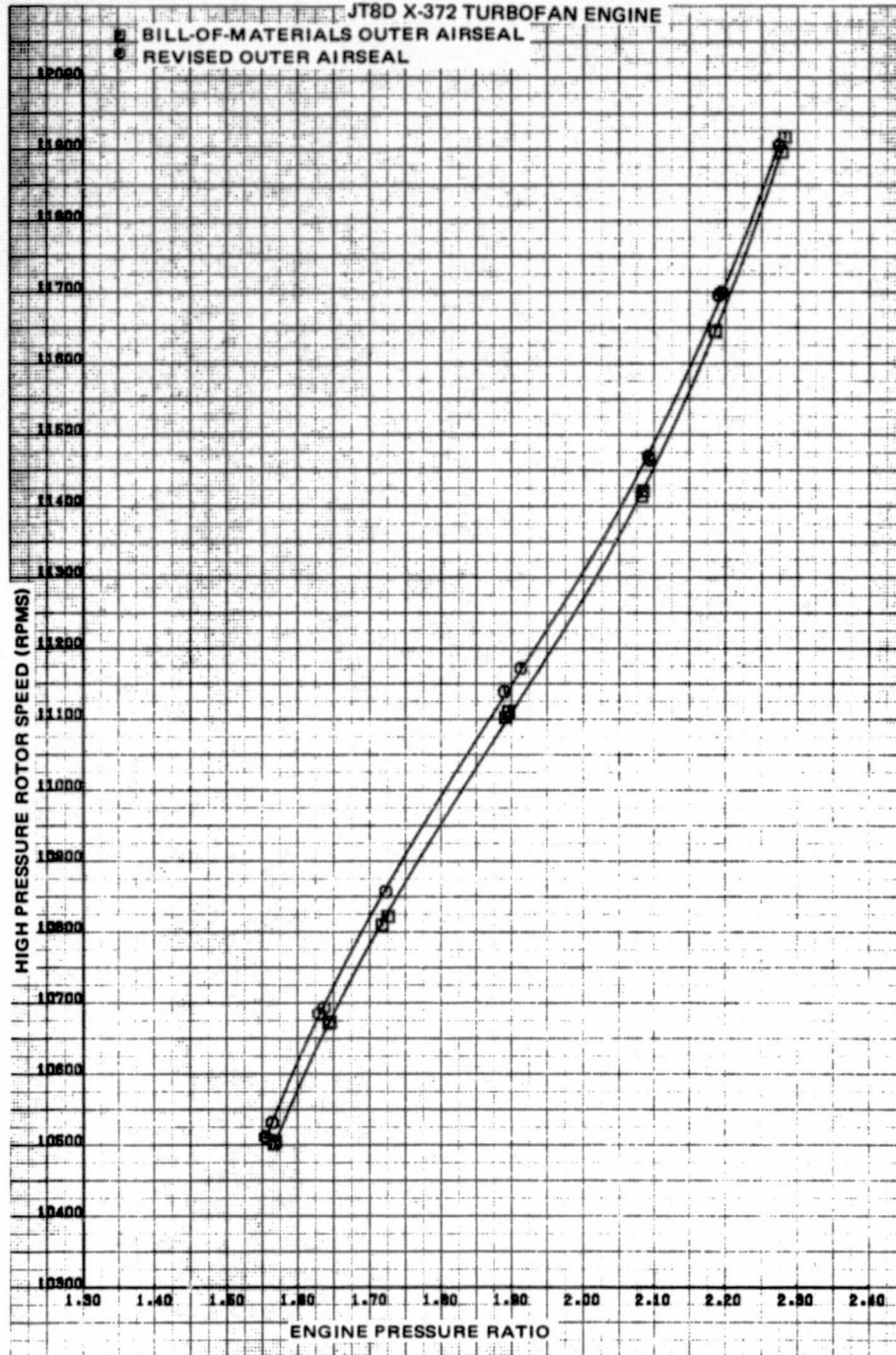


Figure 6-13 High Pressure Rotor Speed Vs. Engine Pressure Ratio for the Bill-of-Materials and Revised HPT OAS at Sea Level Conditions

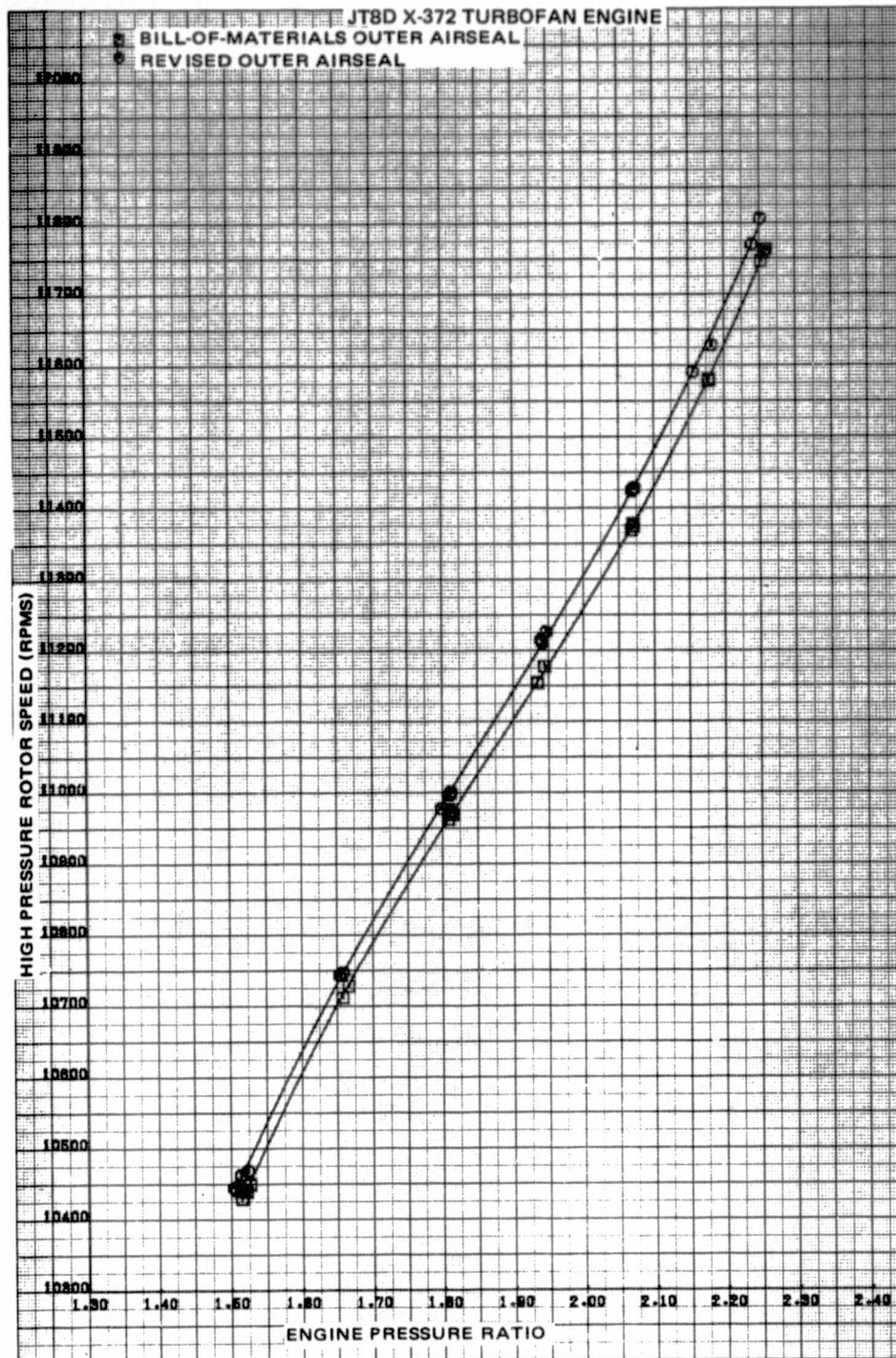


Figure 6-14 High Pressure Rotor Speed Vs. Engine Pressure Ratio for the Bill-of-Materials and Revised HPT OAS at 9,100 m (30,000 ft) Mach Number 0.8 Simulated Conditions

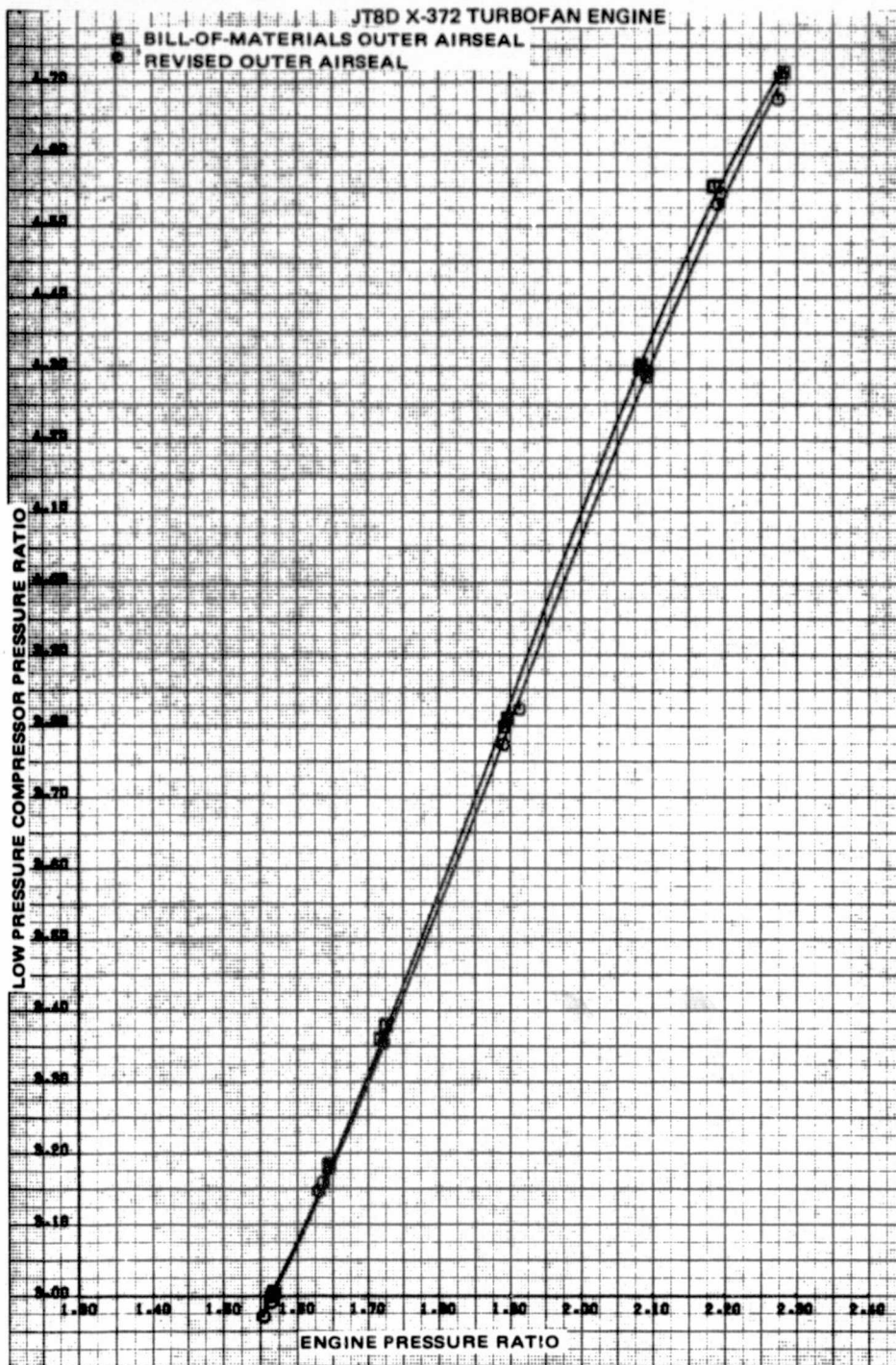


Figure 6-15 Low Pressure Compressor Ratio Vs. Engine Pressure Ratio for the Bill-of-Materials and Revised HPT OAS at Sea Level Conditions

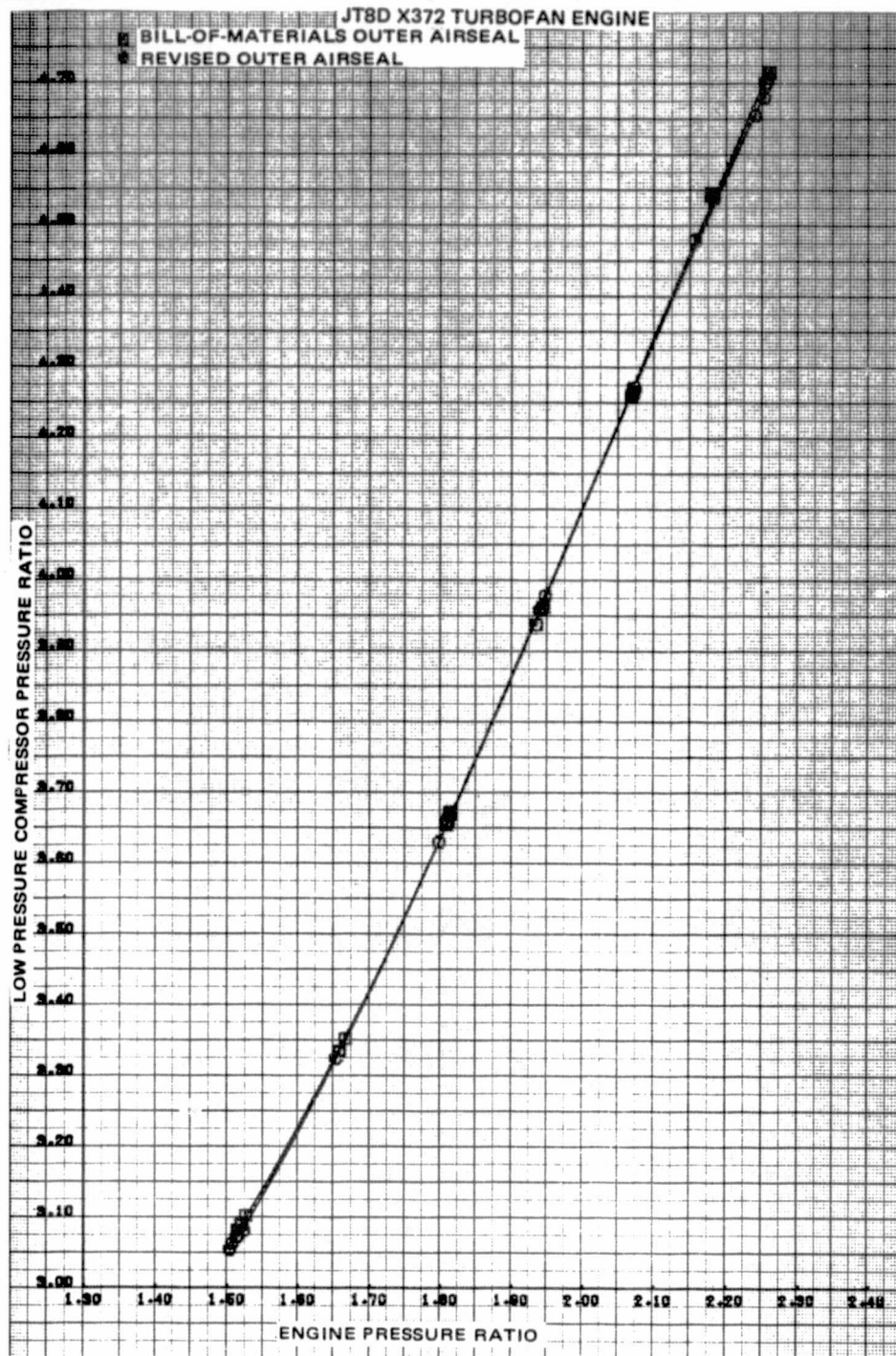


Figure 6-16 Low Pressure Compressor Ratio Vs. Engine Pressure Ratio for the Bill-of-Materials and Revised HPT OAS at 9,100 m (30,000 ft) Mach Number 0.8 Simulated Conditions.

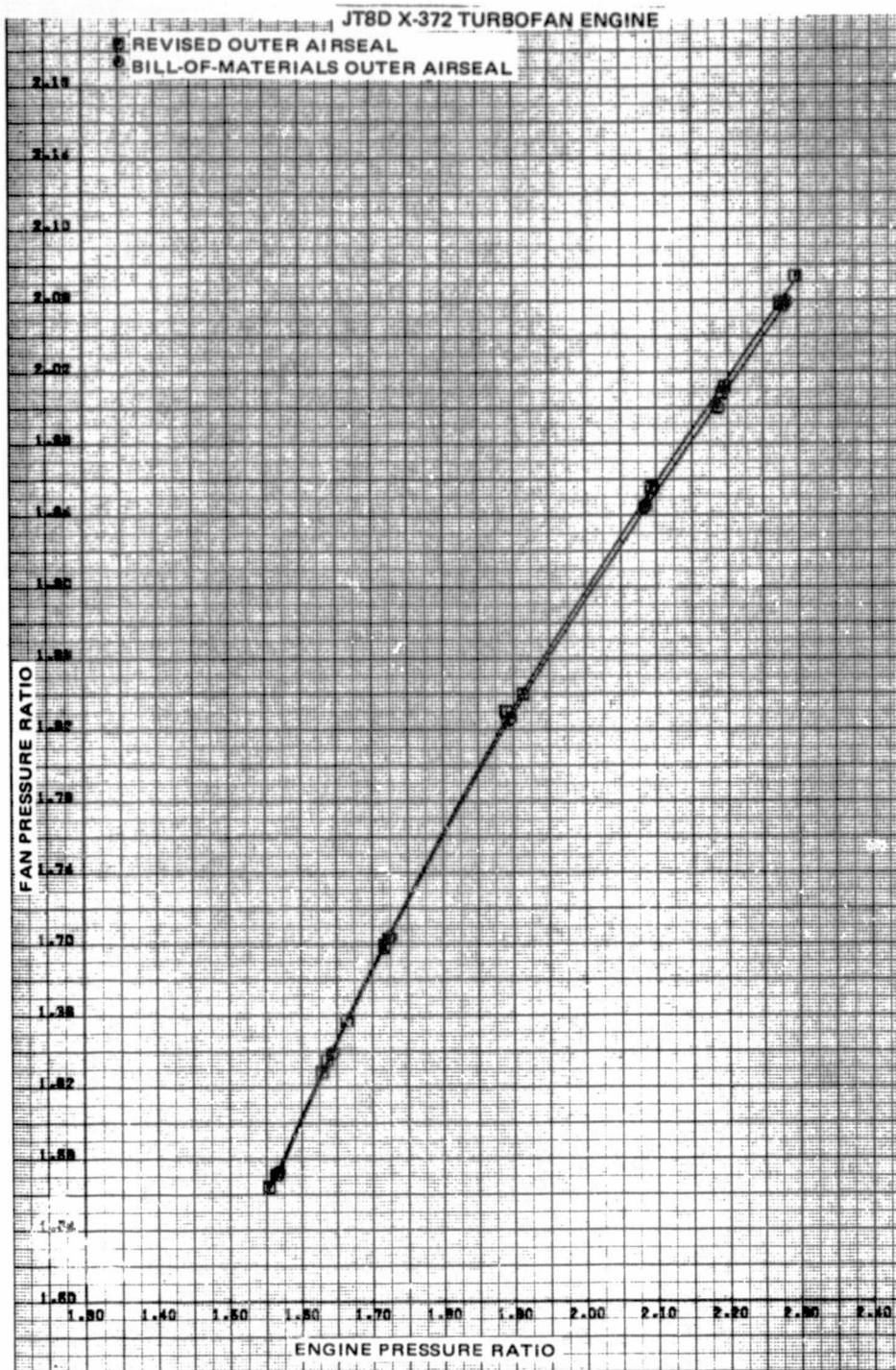


Figure 6-17 Fan Pressure Ratio Vs. Engine Pressure Ratio for the Bill-of-Materials and Revised HPT OAS at Sea Level Conditions

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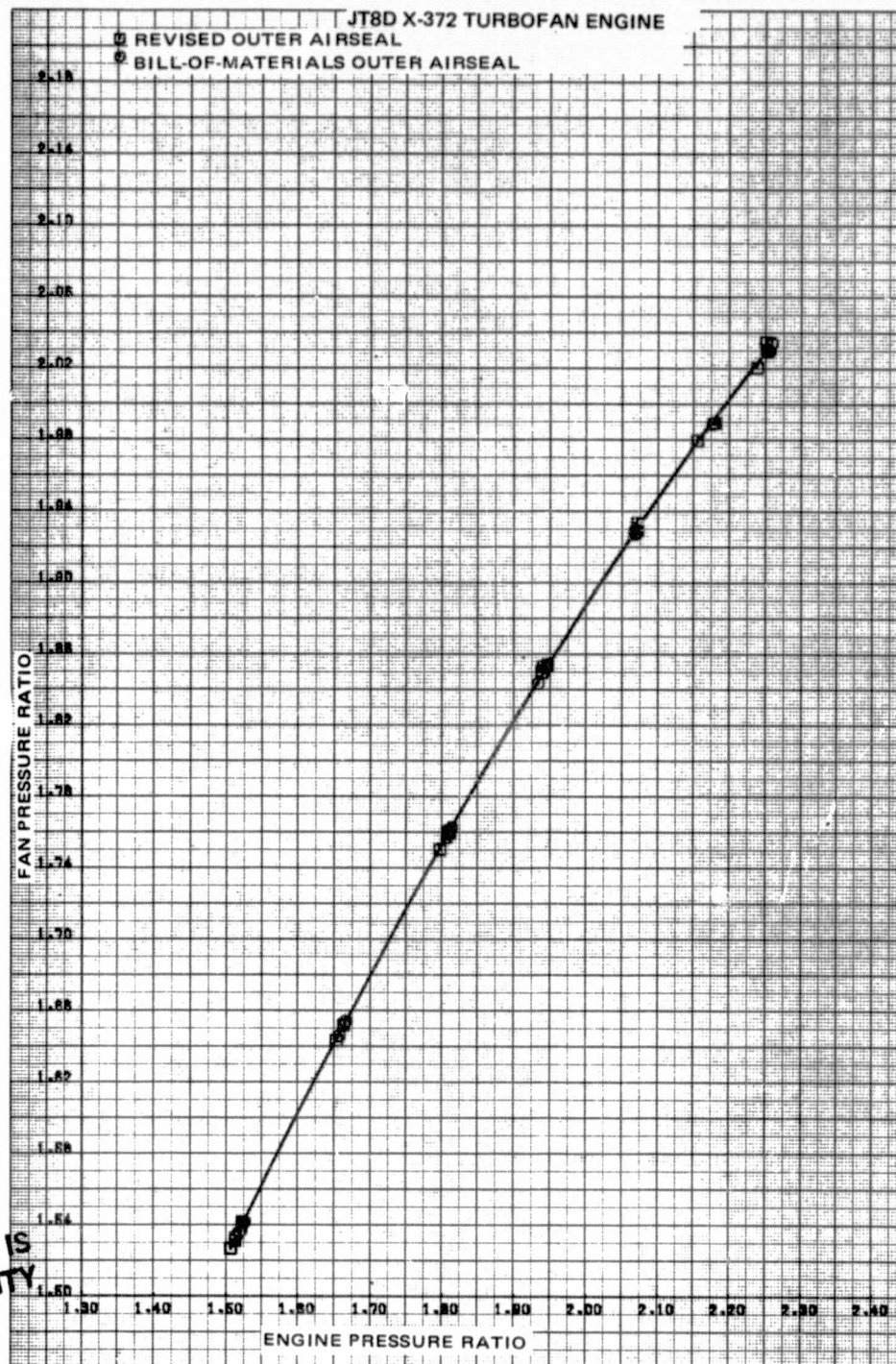


Figure 6-18 Fan Pressure Ratio Vs. Engine Pressure Ratio for the Bill-of-Materials and Revised HPT OAS at 9,100 m (30,000 ft) Mach Number 0.8 Simulated Conditions

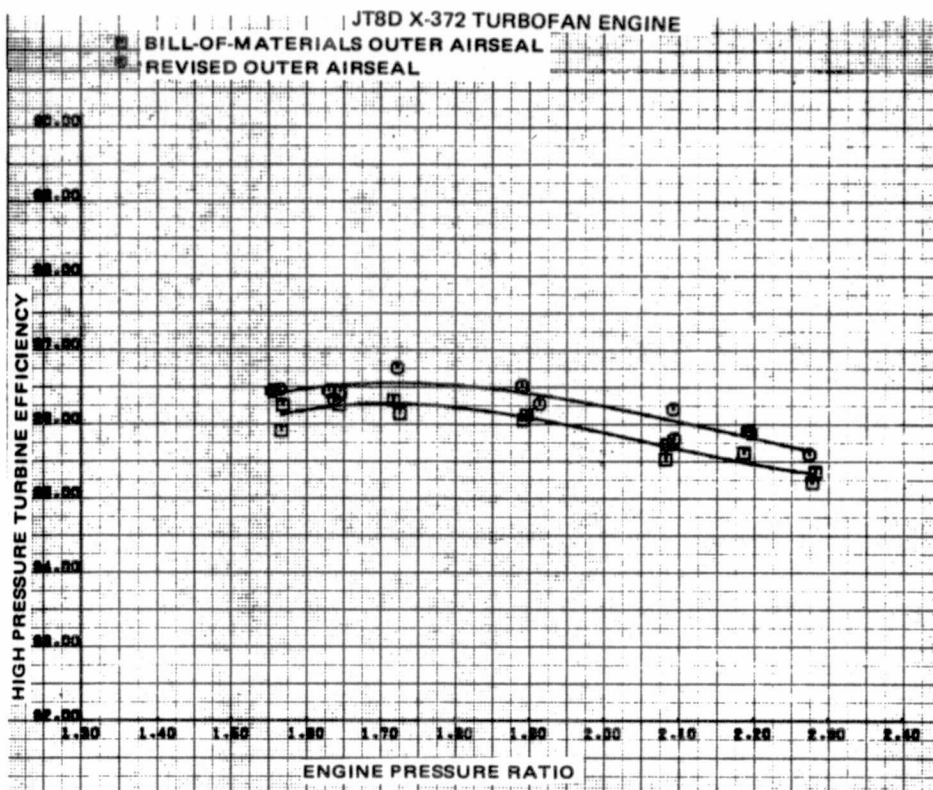


Figure 6-19 High Pressure Turbine Efficiency Vs. Engine Pressure Ratio for the Bill-of-Materials and Revised HPT OAS at Sea Level Conditions.

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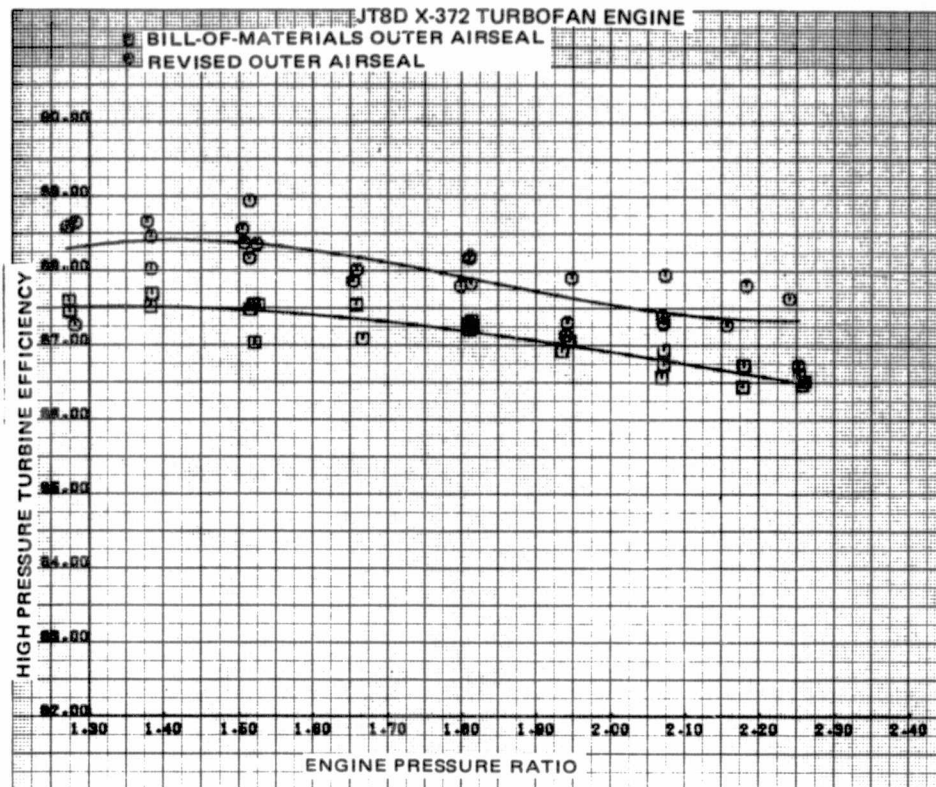


Figure 6-20 High Pressure Turbine Efficiency Vs. Engine Pressure Ratio for the Bill-of-Materials and Revised HPT OAS at 9,100 m (30,000 ft) Mach Number 0.8 Simulated Conditions.

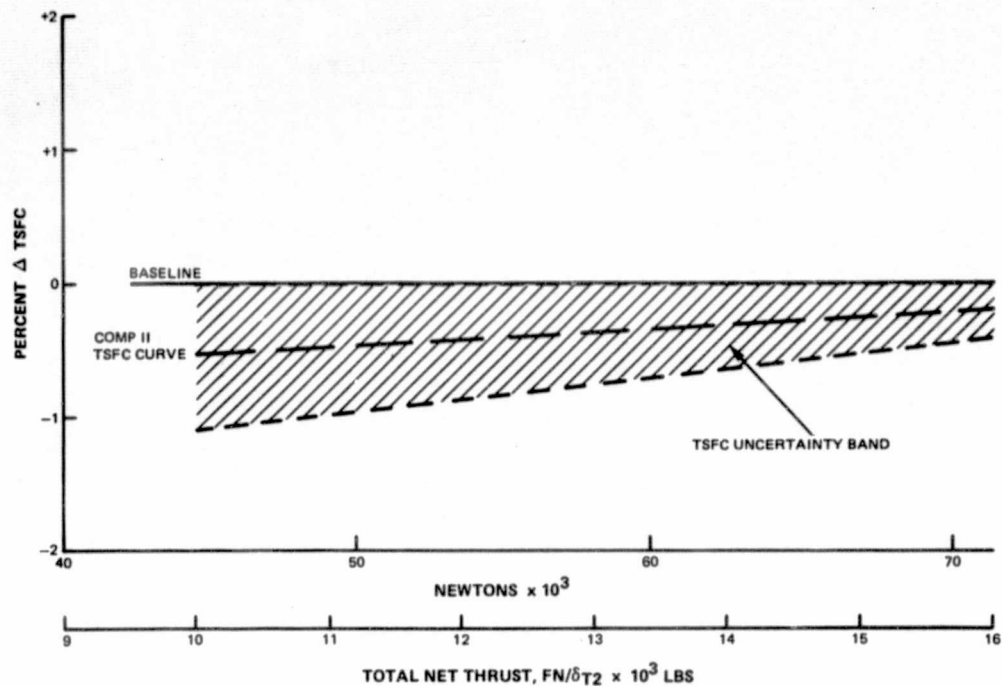


Figure 6-21 Change In TSFC From Baseline Showing COMP II Results and Uncertainty Band At Sea Level Conditions . TSFC shroud gap correction factor is not incorporated.

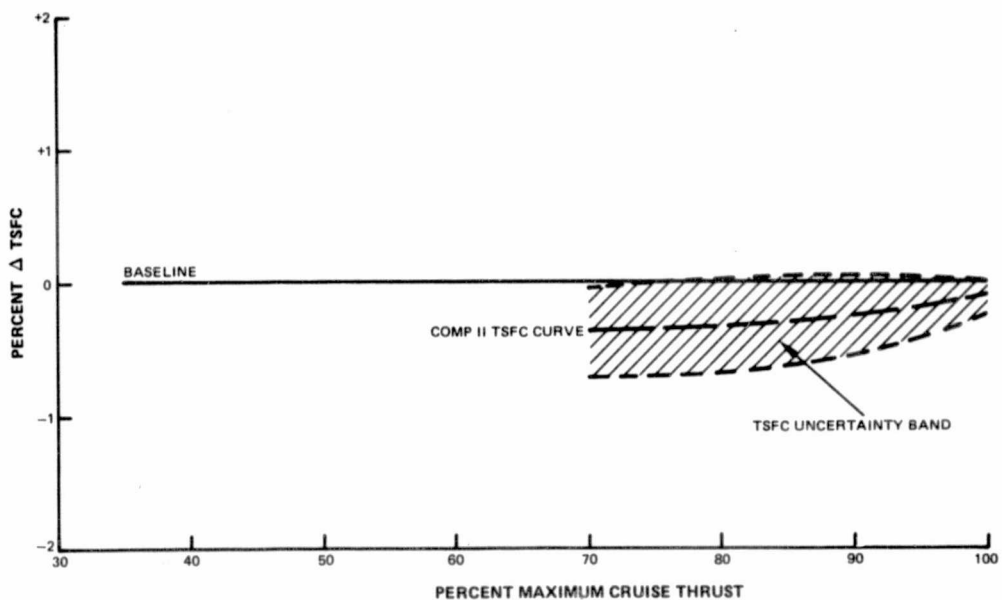


Figure 6-22 Change In TSFC From Baseline Showing COMP II Results and Uncertainty Band At 9,100 m (30,000 ft) Mach Number 0.8 Simulated Conditions. TSFC shroud gap correction factor is not incorporated.

TABLE 6-1
COMP II RESULTS

	<u>Sea Level Takeoff</u>			<u>Altitude 90% Max. Cruise</u>		
	% $\Delta\eta$	% Δ TSFC	Δ EGT °C	% $\Delta\eta$	Δ TSFC	Δ EGT °C
Calculated from Gas Generator Parameters	+0.52 $\pm$ 0.54	-0.21 $\pm$ 0.22	3	0.48 $\pm$ 0.6	-0.24 $\pm$ 0.30	3
Adjustment for Open Shroud Gap	<u>+0.6</u>	<u>-0.24</u>	<u>3</u>	<u>+0.6</u>	<u>-0.3</u>	<u>2</u>
Total	+1.12	-0.45	6	+1.08	-0.54	5

Analytical Approach

A third approach in making a TSFC comparison between the revised OAS and that of the Bill-of-Materials configuration is based on calculated effects of clearance, leakage and mixing pressure loss. These latter parameters are derived from turbine section temperature and pressure data obtained during the engine tests.

The average bulk temperature of the seal was obtained from temperature data generated by thermocouples embedded in the revised OAS. This measurement was used to calculate the radial growth of the seal using the free ring equation:

$$\Delta R_s = R_s \alpha \Delta T_s$$

where R_s = average radius of the seal at 21°C

α = coefficient of thermal expansion of the seal

ΔT_s = bulk temperature change of the seal ($T_{\text{final}} - 21^\circ\text{C}$)

Disk and blade radial growths were calculated using engine test data to scale disk and blade growths from Pratt & Whitney Aircraft thermal models to this particular engine. Clearance of the blade tip to the OAS was calculated by adding the clearance change. The operating clearance for the revised seal at 90 percent maximum cruise-altitude conditions is 0.046 cm (0.018 inches), a reduction of 0.048 cm (0.019 inches) over the Bill-of-Materials seal, which had an operating clearance of 0.094 cm (0.037 inches). The clearance is a result of two effects: the insulative properties of the new seal and a decrease in hot gases flowing past the improved seal, which together allow the seal to operate at a lower temperature.

Seal leakage as a function of clearance was determined using data from a previous cold flow rig test, which was used to generate Figure 6-4. Based on these curves the calculated seal clearances from the engine tests indicate a leakage reduction for the revised OAS of 1.62 percent. This leakage reduction translates directly into a HPT efficiency gain of 1.62 percent at sea level takeoff and 1.58 percent at 90 percent maximum cruise, which would result in a TSFC improvement of 0.65 percent at sea level takeoff, and 0.79 percent at 90 percent maximum cruise.

TSFC should also improve with the new blade cooling air discharge configuration. For the Bill-of-Materials blade there is a 1.5 percent efficiency loss for the 1.5 percent total engine airflow (W_{AE}) discharged at the blade tip. In the revised blade 0.8 percent W_{AE} is discharged at the tip, resulting in only a 0.8 percent efficiency loss, which is a net improvement of 0.7 percent over the Bill-of-Materials blade. The remaining 0.7 percent W_{AE} is discharged through the four suction side holes. The total pressure mixing loss for the suction side discharge was determined using the following equation (Reference 1):

$$\frac{\Delta P_O}{P_O} = - \frac{\gamma M_p^2}{2} \left(\frac{W_c}{W_p} \right) \left[1 + \frac{T_{oc}}{T_{op}} - \frac{2U_c}{U_p} \cos \beta \cos \theta \right]$$

where

- M_p = Local primary gas Mach number
- P_O = Total pressure of primary gas stream
- T_{oc} = Coolant gas temperature at hole exit
- T_{op} = Total temperature of primary gas
- U_c = Coolant mass flow rate
- U_p = Local primary gas velocity
- W_c = Coolant mass flow rate
- W_p = Primary mass flow rate
- β = Angle between coolant gas velocity vector and airfoil surface at point of ingestion
- θ = Angle between the projection of the coolant velocity vector at the point of ingestion and the primary gas stream velocity vector
- γ = Ratio of specific heats

Airfoil surface pressure distributions were generated for the four spanwise locations of the cooling holes to determine flow velocities around the blade section. Figure 6-23 gives a typical pressure distribution. Primary and cooling gas parameters and the mixing losses were calculated for each hole and the results are summarized in Table 6-2. The total pressure loss of 0.502 percent is equivalent to a 0.5 percent efficiency loss, which results in a TSFC penalty of 0.2 percent at sea level takeoff and 0.25 percent at 90 percent maximum cruise.

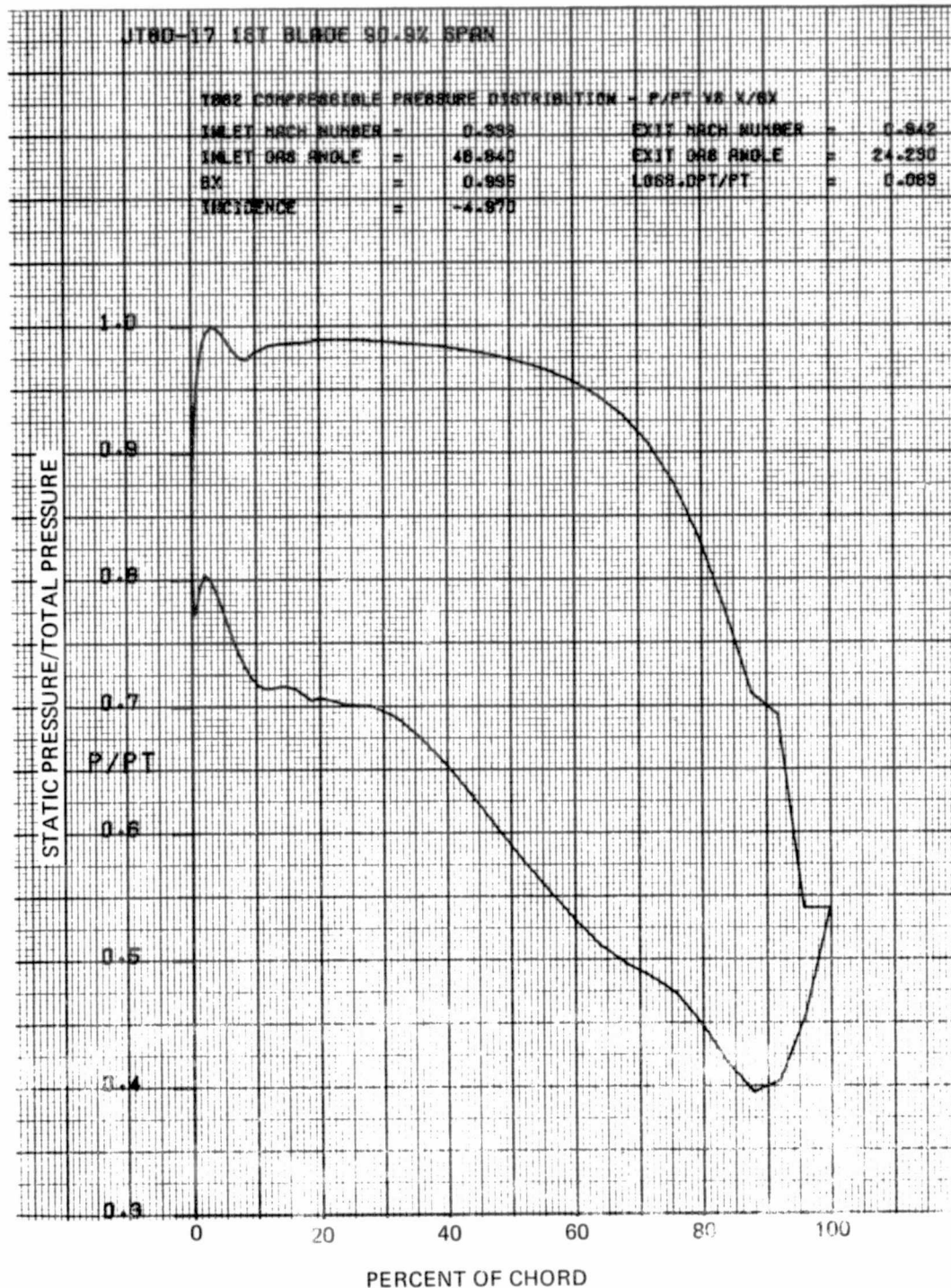


Figure 6-23 Typical Pressure Distribution for the HPT Blade at 90.9% Span.

TABLE 6-2
GAS PARAMETERS FOR THE FOUR SUCTION SIDE
COOLING AIR DISCHARGE HOLES

Hole No.	1	2	3	4	Total
% Span	94.9	90.9	86.6	82.7	
% Chord	44	44	44	50	
M_p	0.894	0.847	0.826	0.865	
W_c — kg/sec (lbm/sec)	0.127 (0.281)	0.127 (0.281)	0.127 (0.281)	0.127 (0.281)	
W_p — kg/sec (lbm/sec)	72.5 (160.0)	72.5 (160.0)	72.5 (160.0)	72.5 (160.0)	
T_{oc} — K (°R)	1033 (1860)	1033 (1860)	1033 (1860)	1033 (1860)	
T_{op} — K (°R)	1229 (2212)	1226 (2207)	1225 (2205)	1223 (2202)	
U_c — m/sec (ft/sec)	156.7 (514)	156.4 (513)	156.1 (512)	155.8 (511)	
U_p — m/sec (ft/sec)	578 (1897)	550 (1806)	538 (1765)	560 (1838)	
β (degrees)	45	46	47	54	
θ (degrees)	0	0	0	0	
$\Delta P_o/P_o$ (%)	0.135	0.120	0.115	0.132	0.502

The calculated efficiency and TSFC effects are summarized in Table 6-3, which shows a net TSFC improvement of 0.73 percent at sea level takeoff and 0.89 percent at 90 percent maximum cruise. The fact that the calculated improvements are larger than the corrected test results (shown on the same table) indicates that the shroud gap correction may be conservative. However, the analytical and measured results are in sufficient agreement to verify the measured data.

TABLE 6-3
ANALYTICALLY CALCULATED RESULTS

	Sea Level Takeoff		Altitude 90% Max. Cruise	
	% $\Delta\eta$	% Δ TSFC	% $\Delta\eta$	% Δ TSFC
Reduced Tip Leakage	+1.62	-0.65	+1.58	-0.79
Reduce Tip Cooling Air Discharge	+0.70	-0.28	+0.70	-0.35
Suction Surface Cooling Air Discharge	-0.50	+0.20	-0.50	+0.25
Total	+1.82	-0.73	+1.78	-0.89
Corrected Test Results		-0.61		-0.60

6.2 ENGINE CONDITION

Hardware was removed at the conclusion of testing to check for abnormalities. None were found. Some localized cracking in the blade tip shrouds occurred during fabrication as described in Section 3.2. These cracks did not propagate during the testing, and had no effect on the test results. A comparison of the Bill-of-Materials and revised hardware after testing is given in Figure 6-24.

6.3 ECONOMIC EVALUATION

The revised OAS concept was evaluated as part of the ECI-PI Task 1 Feasibility Analysis effort (Reference 1) in 1977 to estimate its acceptability to the airline companies and the cumulative fuel saving that would result if it became part of the engine Bill-of Materials. This evaluation was based on analytical estimates of the effects of the concept on engine performance, weight, and cost. These early estimates are shown in the first column of Table 6-4, and the evaluation results are shown in the first column of Tables 6-5 and 6-6. It was on the basis of this evaluation that the revised OAS concept was chosen for demonstration under the ECI-PI program.

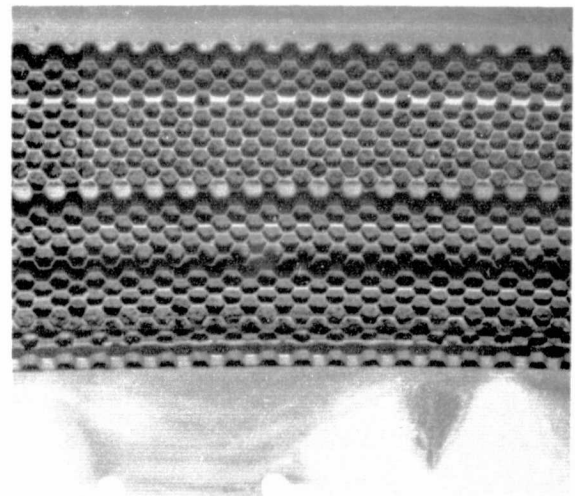
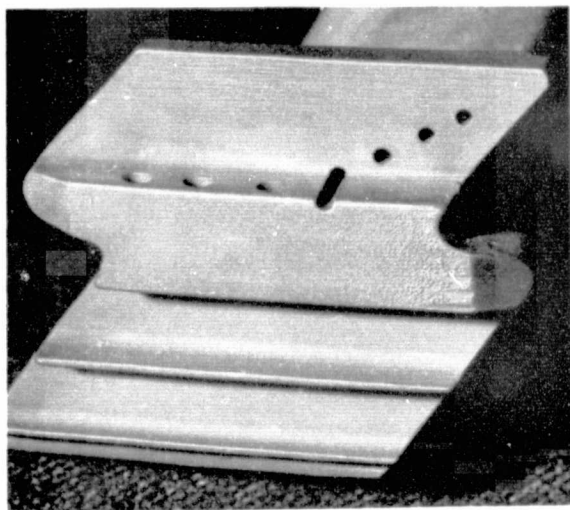
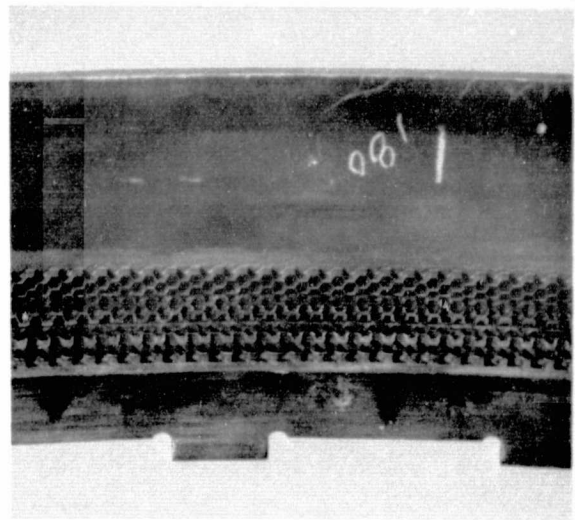
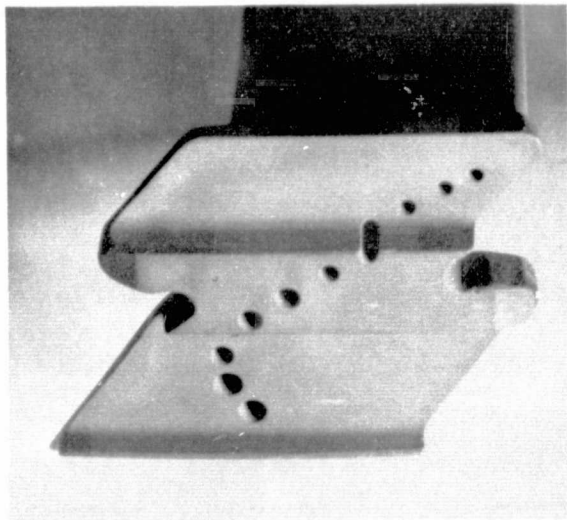


Figure 6-24 Typical Rub Patterns of Bill-of-Materials Blade and OAS, and Revised Blade and OAS Photographs on the top show Bill-of-Materials hardware. Photographs on the bottom show revised hardware.

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TABLE 6-4
PREDICTED ENGINE EFFECTS
JT8D-11, -15, -17, -17R ENGINES (PER ENGINE)

	Original Predictions	Revisions Based On Test Results
TSFC Improvement, %		
Takeoff	0.20	0.61
Climb	0.23	0.30
Cruise, avg.	0.50	0.60
Hold	0	0
EGT Improvement, °C		
Takeoff	4	6
Climb	3	5
Weight Change, kg (lb)	0	
Price Change, \$	+7400	
Kit Price, \$ (attrition basis)	10,500	
Maintenance Cost Change, \$/Oper. Hr.		
Materials	+0.90	
Labor @ \$30 per Man-Hr.	-0.95	-1.24
Start of Service Date	Mid-1980	

TABLE 6-5
AIRLINE COST EVALUATION
727-200 AIRPLANE (PER AIRPLANE)

	Original Evaluation	Revisions Based On Test Results
Total Operating Cost Change, \$/Yr.	-9100	-13240
Required Airline Investment Change, \$		
New Buy	+35,440	
Retrofit	+49,200	
Payback Period, Years		
New Buy	3.9	2.7
Retrofit	5.4	3.7

DC9-50 AIRPLANE (PER AIRPLANE)

Total Operating Cost Change, \$/Yr	-4520	-6740
Required Airline Investment Change, \$		
New Buy	+23,600	
Retrofit	+32,800	
Payback Period, Years		
New Buy	5.2	3.5
Retrofit	7.3	4.9

TABLE 6-6
FUEL SAVING EVALUATION
WORLD FLEET OF 727, 737, AND DC9 AIRPLANES

	Original Evaluation	Revisions Based On Test Results
No. of Engines Affected		
New Buy	720	720
Retrofit	780	2400
Total	1500	3120
Cum. Fuel Saved, 10 ⁶ liters (10 ⁶ gal)		
New Buy	189 (50)	231 (61)
Retrofit	151 (40)	536 (142)
Total	340 (90)	767 (203)
Fleet Fuel Saved, %	0.4	0.5

The second column of Table 6-4 shows the performance effects that were obtained from the engine test program. The takeoff and climb TSFC proved to be significantly better than the early estimates, and the cruise TSFC is slightly better. The EGT improvement proved to be more than the original estimate, which increased the maintenance labor cost saving. Note that the estimates of engine weight, and price effects, and the projected start of service date have not been updated, since the demonstration program provided no information on these parameters.

The results of the economic evaluation are corrected in the second column of Tables 6-5 and 6-6 to reflect the demonstrated performance improvements. The improved payback period for retrofit in existing engines (see Table 6-5) results in a significant increase in the number of engines affected as shown on Table 6-6. This increase, combined the increase from 0.4 to 0.5 percent in the fleet fuel saved estimate, results in a cumulative fuel saving increase from 340 million liters (90 million gallons) to 767 million liters (203 million gallons).

7.0 CONCLUDING REMARKS

Engine testing of the JT8D Revised HPT Cooling and Outer Air Seal concept shows an average cruise specific fuel consumption improvement of 0.6% and a takeoff exhaust gas temperature improvement of 6°C.

The demonstrated performance improvements are better than the estimates used in the Engine Component Improvement Program Feasibility Analysis evaluation.

This better performance will enhance the airline acceptability of the concept and more than double the estimated cumulative fuel saving (as defined by the Feasibility Analysis) to 767 million liters (203 million gallons).

The revised blades and seal showed no unusual wear or degradation following the engine testing.

APPENDIX A

PRODUCT ASSURANCE

INTRODUCTION

The Product Assurance system provided for the establishment of quality requirements and determination of compliance with these requirements, from procurement of raw material until the completion of the experimental test. The system ensures the detection of nonconformances, their proper disposition, and effective corrective action.

Materials, parts, and assemblies were controlled and inspected to the requirements of the JT8D Revised High Pressure Turbine and Outer Air Seal Program. A full production-type program requires inspection to the requirements indicated on the drawings and pertinent specifications. On experimental programs Engineering may delete or waive noncritical inspection requirements.

Parts, assemblies, components and end-item articles were inspected and tested prior to delivery to ensure compliance to all established requirements and specifications.

The results of the required inspections and tests were documented as evidence of quality. Such documents, when requested, will be made available to designated Government Representatives for on-site review.

Standard P&WA Commercial Products Division Quality Assurance Standards currently in effect and consistent with Contractual Quality Assurance Requirements were followed during the execution of this task. Specific standards were applied under the contract in the following areas:

1. Purchased Parts and Experimental Machine Shop
2. Experimental Assembly
3. Experimental Test
4. Instrumentation and Equipment
5. Data
6. Records
7. Reliability, Maintainability and Safety

1. PURCHASED PARTS AND EXPERIMENTAL MACHINE SHOP

Pratt & Whitney Aircraft has the responsibility for the quality of supplier and supplier-subcontractor articles, and effected its responsibility by requiring either control at source by P&WA Vendor Quality Control or inspection after receipt at P&WA. Records of inspections and tests performed at source were maintained by the supplier as specified in P&WA Purchase Order requirements.

Quality Assurance made certain that required inspections and tests of purchased materials and parts were completed either at the supplier's plant or upon receipt at P&WA.

Receiving inspection included a check for damage in transit, identification of parts against shipping and receiving documents, drawing and specification requirements, and a check for Materials Control Laboratory release. Positive identification and control of parts was maintained pending final inspection and test results.

The parts manufactured in Pratt & Whitney Aircraft Experimental Machine Shop were subject to Experimental Construction procedures to ensure that proper methods and responsibilities for the control of various quality standards were followed.

Drawing control was maintained through an engineering drawing control system. Parts were identified with the foregoing system. Quality Assurance personnel are responsible for reviewing drawings issued to vendors to ensure that the proper inspection requirements are met.

Non-conforming articles involving experimental programs are brought to the attention of the test engineer, who may call upon any technical expert he may need to make a disposition on a part. If the test engineer decides to waive or delete a non-critical inspection requirement, the Quality Assurance representative is informed of his decision. This procedure was followed with respect to the excessive shroud gap machining and the cracks associated with the plug welded tip cooling holes.

2. EXPERIMENTAL ASSEMBLY

In Experimental Assembly, an engine was assembled for evaluation of engine performance under the program. Established experimental construction procedures were employed to perform the work and to ensure that proper responsibilities and methods for the control of various quality standards were followed.

3. EXPERIMENTAL TEST

In Experimental Test, performance calibrations were run on the engine. The testing was performed under Experimental Test Department procedures which cover sea level testing in X-16 stand, and altitude testing in X-209 stand.

4. INSTRUMENTATION AND EQUIPMENT

Instrumentation and equipment were controlled under the P&WA Quality Assurance Plan.

The accuracy of gages and equipment used for quality inspection functions was maintained by means of a control and calibration system. The system provided for the maintenance of reference standards, procedures, records, and environmental control when necessary. Gages and tools used for measurements were calibrated utilizing the aforementioned system.

Reference standards were maintained by periodic reviews for accuracy, stability, and range. Certificates of Traceability establish the relationship of the reference standard to standards in the National Bureau of Standards (NBS). Calibration of work standards against reference standards was accomplished in environmental-controlled areas.

Initial calibration intervals for gaging and measuring equipment were established on the basis of expected usage and operating conditions. The computerized gage control system provided a weekly listing of all gages and equipment requiring calibration, highlighting overdue items.

5. DATA

The performance data for the engine in the Experimental test stand was recorded on the Steady State Data System (SSDS), certified to procedures which specified the calibration intervals for the various components requiring laboratory certification. During each data acquisition, the system recorded certified reference parameters which provided an "on-line" verification that the systems were performing properly.

This "confidence" data was reviewed at the time of the run and was later analyzed to provide an overall assessment of the systems operations.

6. RECORDS

Quality Assurance personnel ensured that records pertaining to quality requirements were adequate and maintained as directed in Experimental Quality Assurance procedures and in accordance with contractual requirements.

Engine build and operating record books were maintained in accordance with FAA approved practices. In addition, a consolidated record of operating times for each major engine component used in experimental programs was kept.

An equipment log for performance improvement vehicles was maintained in accordance with the P&WA Quality Assurance Plan.

7. RELIABILITY, MAINTAINABILITY AND SAFETY

Standard production engine design techniques and criteria, which consider product reliability and maintainability in context with all other requirements (such as performance, weight and cost), were used in defining the modified high pressure turbine blade and outer air seal for the Revised Outer Air Seal Program. Critical stress and temperature areas of the modified parts were analyzed to insure that their structural margins are equal to or better than those of the Bill-of-Materials parts. This expectation would be verified in an engine development and certification program before the modifications are released to production.

The experimental modified blades used in the Revised OAS demonstration testing were fabricated from semifinished production castings, using non-production welding and machining operations. Local cracking was experienced around some of the welds as described in Section 3.2, but the cracks had no detrimental effects on the project objective, which was to demonstrate the performance improvement of the concept.

The safety activities at Pratt & Whitney Aircraft are designed to fully comply with the applicable sections of the Federal Aviation Regulations, Part 33 Air Worthiness Standards: Aircraft Engines, as established by the Federal Aviation Administration."

REFERENCE

1. Shapiro, Archer H. – The Dynamics and Thermodynamics of Compressible Fluid Flow, Volume I, Ronald Press Co., N. Y. 1953.
2. W. O. Gaffin and D. E. Webb – JT8D and JT9D Jet Engine Component Improvement Program – Task 1 Feasibility Analysis – Final Report. NASA CR-159449